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Project Ares

A Systems Engineering and Operations Architecture for the Exploration of Mars

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John M. Amrine, Jeff M. Berger, David J. Blaufuss, Richard W. Boltz, Michael T. Dunn, Jeffrey S. Gruner, Benjamin C. Huff, George R. Kather

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13. ABSTRACT *(Maximum 200 words)*

The authors, a group of 18 veteran space operations officers, explored the concept of placing a permanently manned station on Mars – Project Ares. Using a five phase plan involving an early moon base and three scientific missions to Mars followed by cargo missions and eventually manned trips to the planet, the authors analyzed, designed, and documented their approach. The report focuses on the second phase of the project – scientific expansion of our knowledge of Mars and proof of a reliable transportation system. Among the subjects discussed in the report are management, mission profile and environment, and mission systems. The latter subject contains a variety of systems proposed by various authorities in the space industry and selected to fulfill the unique requirements of the project. Because this phase is an integral part of the project, emphasis was on reusability and versatility.

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FOREWORD

Man has historically desired to venture beyond his immediate surroundings and explore new frontiers. From the advent of the liquid-fueled rockets to space shuttles and aerospace vehicles, the goal to discover new and different worlds is now becoming a reality.

> ...Our goal is to build on America's pioneer spirit and develop our next frontier...space. Nowhere do we so effectively demonstrate our technological leadership and ability to make life better on Earth...Our progress in space–taking giant steps for all mankind–is a tribute to American teamwork and excellence...We can follow our dreams to distant stars, living and working in space for peaceful, economic, and scientific gain.

President Ronald Reagan
State of the Union Address
25 January 1984

> The first generation of space-probes has revealed so much strangeness and grandeur on our neighboring worlds that, as soon as the technical means are available, men will certainly visit them. And having done this, they will find reasons for living there–as on this earth they have established themselves in such improbable spots as the South Pole and the bottom of the ocean. Space itself, to the considerable surprise of most people, has turned out to be a benign environment; it is only the planets that are hostile. Even this was anticipated almost a century ago by Tsiolkovsky; he regarded the weightless realm of space as man's ultimate home...The escape from gravity which we have all known in our dreams may remind us of life's origins in the ocean; but it may also anticipate a far longer future in space. Cosmonaut-artist Alexei Leonov, the first man to step out of a space capsule and to float in the void, entitled his book of paintings: *The Stars are Waiting.*

Arthur C. Clarke
Sri Lanka
1980

I. Executive Summary

1.1 Project Objectives

The objectives for the GSO-92D design team are to:

1. Conduct preliminary research into a space-related systems engineering problem utilizing a project management approach.
2. Intelligently analyze the support technologies and space hardware necessary to fulfill mission requirements.
3. Understand the trade-offs and complexities involved in planning for space missions.

1.2 Mission Statement

Plan Phase II of the **Project Ares** program: Analyze, design, and plan for scientific research missions at mission requirements and hardware subsystem levels which support the ultimate goal of establishing a permanently manned station on the planet Mars.

1.3 Mission Introduction

The ultimate goal of Project Ares is to establish a permanent manned presence on Mars. To accomplish this goal, a five-phased approach has been adopted using a ***crawl-walk-run*** philosophy in the planning of the Project Ares missions. Accordingly, each phase is an incremental step in supporting the final goal. Since the class project is to develop Phase II of Project Ares, the specific details of Phases I, III, IV, and V have been omitted from this report. A summary description of the entire Project Ares program and macro-level schedule are incorporated for completeness (reference Figure 1).

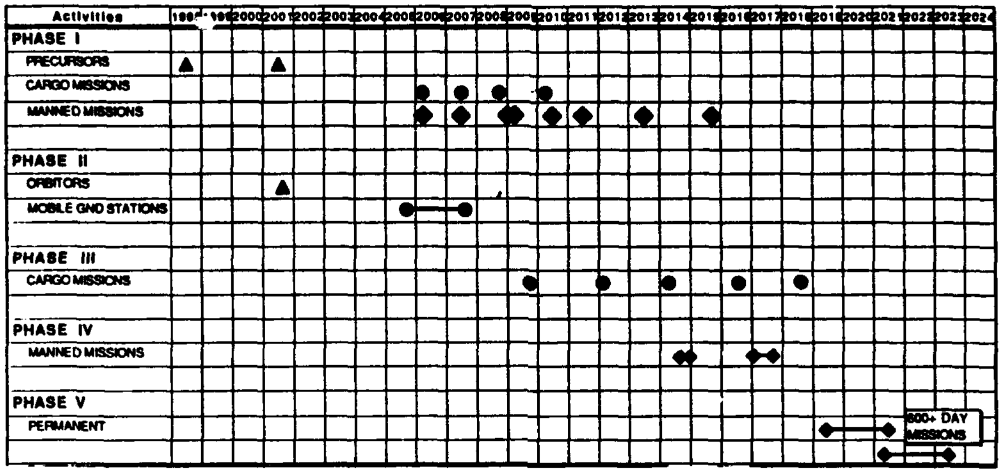

Figure 1. Project Ares Master Schedule.

The first manned mission to Mars is scheduled to occur in 2014 to take advantage of the optimal synodic relationship of the planets that occurs once every 15 years.

1.3.1 Phase I: Project Luna. Man's permanent presence on the lunar surface is a basic foundation of Project Ares. Although the scope of Project Luna will be extensive in terms of observatory operations, scientific research, et cetera, we will only discuss those activities which support (and are considered essential) to Project Ares. These activities are grouped in the Lunar Test Bed Program (LTBP) operating within Project Luna. Since the main objective of Phase I was to develop, test, validate, and refine the procedures and hardware designs to be used on Mars, our concept will use the moon as a risk reduction test-bed.[1] Specific technologies and operational considerations to be conducted on the moon in support of the Mars mission are discussed below.

1.3.1.1 Transportation. A major aspect of the transportation system validation will be the verification of the configuration of the payload. Knowledge gained from this experience will help optimize the payload proportions for the Mars missions. The propulsion system to be used for the missions to Mars will be validated and tested to include the initial use of the Mars Transfer Vehicle (MTV).

1.3.1.2 Habitation. The Mars habitat will be completely designed and *field-tested* as a baseline habitat for the Moon. This concept will provide years of operational experience along with reliability and maintainability data prior to first use on Mars. The habitat is envisioned to arrive on a cargo vehicle ahead of the manned mission and deorbit to the Lunar surface. Once the habitat is employed on the Moon, a nuclear power source and emergency photovoltaic backup capability will be evaluated and refined. The habitat will be designed to operate as a *closed system*. Its continuous use on the Moon will yield valuable data to support the perfection of the *closed system* concept. All aspects of autonomous life support and environmental control systems, and their back-ups, will be monitored and improved based upon the Moon habitat experience.

1.3.1.3 Operational Capacity. Very few capabilities necessary for Mars missions cannot be rigorously tested and refined on the Moon; therefore, one objective of Phase I is to test every feasible system and technology that will be required on Mars. For example, some capabilities such as the generation of breathable gases from the CO_2-rich atmosphere and the Martian soil are not feasible on the Moon, since it has no atmosphere. Most required capabilities on Mars can (and will) be replicated to some degree on the Moon. Some of the capabilities to be fully explored on the Moon are:

---

[1] We assume that Project Luna activities will be conducted on a timeline which supports the Project Ares mission schedule.

- Research life support self-sufficiency to include air, water, and food generation experiments.
- Develop a maintenance capability.
- Develop a construction capability.
- Develop and expand a teleoperation and telepresence capability.
- Conduct scientific experiments.

1.3.1.4 Life Sciences. The goal of the life sciences portion of the LTBP is to collect an "encyclopedic" wealth of data to support design of systems enabling man to survive the interplanetary journey from Earth to Mars, an extended stay on the Martian surface, and the return trip back to Earth. Project Luna activities using Mars-designed equipment can be conducted to accomplish both their primary mission and the secondary mission of life sciences data collection for Project Ares. A macro list of the promising areas for scientific exploitation include, but are not limited to:

- Psychological evaluation of the effects of confinement during interplanetary travel and extended absence from Earth, interpersonal relations among personnel living and working in close quarters, and investigation of living habitat design to alleviate the long-term effects of spartan, functional living spaces on crew morale and productivity.
- Physiological evaluation of the effects of long-term exposure to the radiation and weightless environments of space and space travel. Here data collection may center on calcium loss in bone tissue and subsequent changes in bone density, cardiovascular fitness and changes in the heart muscle, changes in

blood chemistry, changes in the human immune system, motion sickness, and cell growth when exposed to high levels of cosmic and galactic radiation.

- Investigation of the long-term effects of exposure to zero and micro-gravity environments (Lunar gravity is approximately 0.16 that of Earth; Martian gravity, approximately 0.38).
- Investigation of systems design for hygiene and waste management/waste recycling systems. Operation and evaluation of both travel-based and in situ systems will benefit Project Ares.

1.3.1.5 Resource Generation, Location, Production, and Storage. The ultimate goal of the resource generation, location, production, and storage portion of the LTBP is to build toward a demonstrable level of self-sufficiency for a limited human population on Mars. There are four broad areas of interest in Project Luna that will benefit Project Ares: 1) food production systems and techniques, 2) validation of remote sensing equipment and techniques for the location and quantification of life sustaining resources, 3) remotely controlled materials gathering and processing equipment for the extraction and storage of gases and minerals, and 4) transfer of extracted minerals and gases to either storage containers or operating systems.

Since we begin our Mars efforts with an austere operating capability and tenuous supply line, it is essential that we maximize the accomplishments of our first missions in the area of resource self-sufficiency. Early checkout and validation of remote sensing equipment and techniques on the Moon will allow for their use in the reconnaissance stage of Phase II. With knowledge of where resources vital to supporting human life are located on Mars, we can make wiser choices concerning Mars base site locations. Having located and quantified sizable deposits of minerals essential to supporting life on Mars, we will turn our efforts to the extraction, storage, and ultimate transfer of these minerals to the life supporting systems that will consume them. Food irradiation holds out promise for sending stores of fresh food to Mars, but life there cannot be completely dependent upon supply ships. In order to establish a true measure of self-sufficiency, man must learn how to grow and cultivate food in the Martian environment.

1.3.2 Phase II: Unmanned Scientific Exploration of Mars. This report principally considers the Phase II objective to provide a detailed exploration of Mars in support of a site selection decision for a permanent manned presence. Meeting this objective will require the placement of remote sensors in Martian orbit and surface monitoring probes and robotic surface entities on the Martian planet. Since manned presence will eventually be permanent, the selected site must optimize the availability of usable natural resources. For detailed information on mission requirements, refer to Section 1.6. The master schedule of Phase II activities is provided in Figure 2.

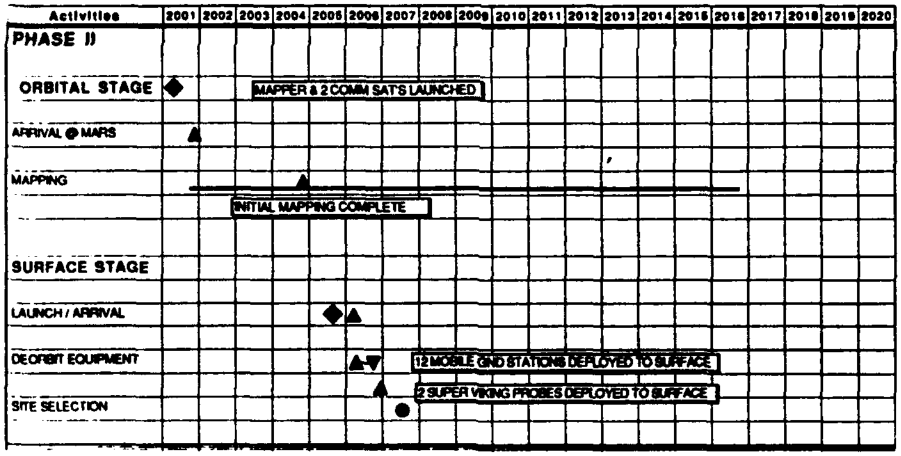

Figure 2. Project Ares Phase II Schedule.

1.3.3 Phase III: Pre-Staging of Equipment. Phase III will pre-stage equipment and supplies and establish the infrastructure necessary to support human life in the Martian environment. Phase III will consist of *cargo only* flights into Martian orbit and the deorbit of the habitat to the Martian surface. After deorbit, the habitat will be remotely monitored to ensure proper operation of the nuclear power plant and life support systems. During this phase, a third and fourth communications satellite will be transported on the cargo missions. The third communications satellite will augment the two already in Mars synchronous orbit, thus providing full coverage in preparation for manned presence. The fourth satellite will serve as a spare that will be activated when needed. Specific details of Phase III are not developed in this paper.

1.3.4 Phase IV: First Manned Presence. Phase IV will be initiated with the first manned mission to Mars which will last 30–60 days. The objective of the first mission is to bring the Martian base up to a fully operational status to include breathable gas and potable water generation. At the conclusion of this *set-up* mission, the habitat will be ready for long-term occupancy and will be placed in an autonomous minimum operating condition. Subsequent missions will expand the base of operations with additional habitat modules, thereby incrementally increasing the on-station time to 180 days. Specific details of Phase IV are not developed in this paper.

1.3.5 Phase V: Permanent Manned Presence. Phase V begins with the first 600-day mission on the Martian surface. Due to orbital mechanics constraints, minimal transit times (which minimize exposure to cosmic and galactic radiation) only occur once every 26 months. Phase V is based on the assumption that manned missions will be conducted during these windows of opportunity which will leave each mission crew on the Martian surface approximately 600 days. The 600-day missions will be continually expanded until a fully autonomous Martian populace is established. Specific details of Phase V are not developed in this paper.

1.4 *Mission Objective*

We propose to plan Phase II of Project Ares to include the development of mission requirements and preliminary design of selected hardware. Phase II will continue scientific data collection about Mar's surface and atmosphere with the goal of expanding man's knowledge of the planet to aid the process of selecting landing sites for both manned and robotic exploration. Areas of scientific interest include:

- Martian surface mapping.
- Solar radiation measurements.
- Chemical and mineral analysis.
- Soil toxicity analysis.
- Volcanic and tectonic activity.
- Polar region analysis.
- Atmospheric composition analysis.
- Weather data collection.

Phase II wil! also begin the establishment of a Mars-to-Earth communications capability and a rudimentary Mars surface navigation capability.

1.5 *Mission Profile*

1.5.1 Phase II, Orbital Stage. The MTV will place a Mars Surface Mapper (MSM) and two Mars Communications Satellites (MaRCoS) into Martian orbit. The mapper will enter a sun synchronous orbit and begin recording surface images and collect data on the upper Martian atmosphere. Design teams will investigate means of transmitting the stored data back to Earth. From the data collected during this part of Phase II, scientists will select twelve candidate sites for landing Mars exploration equipment.

1.5.2 Phase II, Surface Stage. This stage will involve sending more scientific equipment to Mars and placing some of it on the Martian surface for further data collection and landing site selection and certification. Scientific equipment will provide the capability to collect weather data from surface stations and relay the collected data back to Earth via the communications satellites placed during the Orbital Stage.

1.6 Mission Requirements

Specific mission requirements to accomplish in Phase II are:

- Develop the requirements and specifications for an interplanetary transportation system. The MTV will carry orbital and surface payloads to Mars. The initial MTV need not be man-rated, but should be developed with that end in mind. The transportation system will support various types of payloads during transit and deploy them into required orbits or descent trajectories. The MTV design will consider requirements for future Project Ares phases, not just Phase II.
- Develop the requirements and specifications for the MSM that will conduct high resolution, multispectral imaging to accurately map the planet's surface. Auxilliary payloads aboard the spacecraft will be scientific instruments to gather strategic data for use in landing site discrimination and selection and to study the Martian atmosphere and weather systems. Photographic resolution requirements are estimated to be on the order of one meter per picture element (pixel).
- Develop the requirements and specifications for a Martian timing and navigation system that is both self-controlled and self-sustaining. The Martian timing and navigation system will have a secondary detection and warning mission for solar flare activity.

- Develop the requirements and specifications for a communications relay capability to provide data relay from reconnaissance spacecraft and scientific data collection equipment at Mars back to Earth.
- Plan the collection and transmission of scientific data from Mars back to Earth. Data collection should include, but not be limited to, atmospheric, hydrological, meteorological, tectonic, volcanic, and soil chemical and toxicity testing.
- Plan for the investigation of the effects of long-term space exposure on mission equipment enroute to Mars and discuss the possibility of either manned or robotic excursions to the Viking landing sites to investigate the effects of long-term exposure to the Martian atmosphere.

1.7 Critical Assumptions

In order to make the scope of Phase II of Project Ares manageable, it was necessary to make certain key assumptions. These assumptions were deemed necessary to make the overall mission plausable and to limit its scope.

1.7.1 No Budget Constraints—No Cost Accounting. Current cost projections for the Space Exploration Initiative (SEI)[2] exceed $400 billion, but Congressional support for SEI is questionable at any cost. Of the $37 billion sought by the Bush Administration for NASA's 1991 SEI program budget, Congress appropriated nothing (9:25). According to John Logsdon of the publication *Ad Astra*, Congress did not kill the idea of SEI, but simply refused to fund it. Congress stated that they "deferred consideration...[of SEI]...due to severe budget constraints," but added that "it is implicit in the conduct of the nation's civilian space program that such human exploration of our Solar System is inevitable" (58:38). To date, no agency has performed a thorough enough study to adequately predict the actual cost of sending men to Mars, primarily due to the budget uncertainties of needed advances in technology.

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[2] On the 20th anniversary of the Apollo 11 moon landing, July 20, 1989, President Bush proposed a plan to establish a set of long-term goals to give U.S. space programs a purpose and direction. SEI is the first step to sending Americans " back to the Moon. Back to the future. And this time back to stay. And then a journey into tomorrow — a journey to another planet — a [human] mission to Mars" (30:16-23).

We assume that the political, budgetary, and public support necessary to achieve a project of this magnitude do, in fact, exist. This assumption allows the class to focus on the project objectives and engineering design concepts. Accordingly, no budget constraints will be imposed on the design teams.

1.7.2 Space Station Freedom (Modified for the Space Exploration Initiative) Availability. The National Aeronautics and Space Administration (NASA) has always envisioned using Space Station Freedom as a stepping stone for interplanetary travel. Indeed, the initial designs of the space station encompassed a *dual keel*[3] arrangement specifically to meet the servicing and repair operations of satellites and space transportation vehicles with minimal disturbance to sensitive microgravity and materials processing procedures on-going in the scientific modules. Assembly, servicing, and processing operations of the Mars Transfer Vehicles and Mars Excursion Vehicles were planned for Freedom. A dual mission processing facility, for concurrent Lunar and Martian missions, would be located in the lower keel location of the space station(88:6-7).

Specifically, Space Station Freedom would provide the following capabilities (2:3):

- Serve as a laboratory for determination of acceptable long-term human spaceflight microgravity and radiation countermeasures.
- Be a source of technology, hardware, and software for Lunar and Martian vehicles and systems.

---

[3] Two long and parallel interconnected structures similar in concept to a twin-hulled sailboat design.

- Serve as a testbed for validation of Lunar and Martian systems and technology elements.
- Accommodate the assembly, test, payload mating, launch, refurbishment, and refueling of Lunar and Martian vehicles.

Unfortunately, during the mid-1980s NASA eliminated the dual keel design with hopes to gradually recoup this capability through evolution in the late 1990s. That desire is now essentially unattainable since Congress (October 1990), following recommendations from the Advisory Committee on the Future of the US Space Program, charged NASA to completely redesign the station, thereby reducing its complexity and costs. Needless to say, this *restructuring* effort further removed any SEI capabilities in the foreseeable future. This fact was clearly recognized by the National Research Council in its report to the National Space Council, even before the restructuring effort, where it stated "...the Space Station is an integral first step...but its present design may not meet all the requirements...[of SEI]." NASA, to date, will not speculate when—after the year 2000—Freedom will evolve to a useful SEI configuration, nor what that configuration will be (28:22).

For the purposes of this project, Space Station Freedom, or its equivalent, will be configured and available for on-orbit assembly and servicing of Mars mission spacecraft. This requirement is necessary for three reasons. First, space shuttle resources are extremely limited and costly to operate. Second, this concept of operations reduces the risk associated with astronaut extra-vehicular activities by using space station telerobotics capabilities. Finally, costs and complexity of operations are reduced through on-orbit operations versus overcoming Earth's gravity with direct ascent launches.

1.7.3 National Launch System (NLS) Availability. The NLS grew out of the lack of and need for a heavy lift launch vehicle to support the requirements of the Earth-orbiting space station and SEI, as well as for the need of a complementary system to the space shuttle. Originally established in 1987 as the NASA-USAF Advanced Launch System (ALS), the system was designed to place payloads up to 220,000 pounds into low-Earth-orbit (LEO). Through a series of budget restructuring and renaming efforts, ALS became NLS in 1992 with over half of its jointly requested $300M budget cut by Congress (94:14).

For Project Ares, we assume the NLS is operational and capable of lifting 250 metric tons into LEO for rendezvous and docking with an Earth-orbiting, manned space station. The NLS should already be operational as part of the infrastructure necessary to place SEI elements at the space station and support the launching of modules for the Project Luna program. The space station and Project Luna programs and infrastructure are precursors to the five phases necessary to establishing a permanent manned presence on Mars and, therefore, should all be mature before we begin our operations towards Mars. The NLS has also been deemed necessary to meet Vice-President Quayle's request that the United States space launch community accomplish America's space exploration goals of "faster, cheaper, safer, and better" than current systems allow (85). The space station SEI elements and Project Luna components could be launched with a lesser capable vehicle—for instance, one capable of only lifting 150 metric tons to LEO; however, more launches and assembly of equipment in Earth orbit would be required, thus violating the *faster, cheaper, safer, and better* stipulations that can be met by a 250 metric ton capable NLS. We recognize this resource constraint; therefore, we will restrict our maximum component weight to 250 metric tons and assume all Project Ares missions start from an assembled spacecraft in Earth orbit. In Chapter IV, we take a cursory look at the support requirements that Phase II will impose on the NLS.

1.7.4 Space Transportation System (STS) Availability. The Space Transportation System will be available to support launch operations to the Earth orbiting space station. This capability is required to support our design philosophy which calls for the assembly of the Mars transportation vehicle near the space station and the initiation of Mars missions from Earth orbit. We have, in effect, *assumed away* the logistics problem of getting the spacecraft components and payloads into Earth orbit, assembled, integrated, and tested. Although this is a monumental task in and of itself, it is a foreseeable evolution of our current technologies.

1.7.5 No Parasitic (Piggy-Back) Payloads. In today's highly competitive market with constrained launch as well as fiscal resources, private industry and the government typically populate a satellite vehicle with numerous *parasitic* payloads. Examples of current DoD spacecraft which harbor parasitic payloads are the Defense Meteorological Satellite Program and the Global Positioning System. Add-on requirements are usually unrelated to the primary mission of the spacecraft, but do use the limited systems capabilities of the spacecraft bus, such as electrical power, communications bandwidth, et cetera. Because of the technical complexities of the Mars missions and the limited amount of research time afforded this project, no parasitic submissions will be considered.

1.7.6 Parallel Efforts Underway to Validate Needed Mars Technologies. The LTBP is under way by another program office within the Project Luna program to lay the ground work for the long-term habitation of Mars. The LTBP activities will establish the infrastructure necessary to support a manned presence on Mars, and validate the technology, equipment, and operating procedures required for the mission to Mars. Non-LTBP data collection will be conducted to determine the effect of long-term space exposure on man-made hardware and systems by returning to one of the six Apollo landing sites and examining some of the hardware left there. The LTBP activities will also serve as rehearsals for the mission to Mars, while at the same time, gathering and returning more exploratory data from the Moon and acquiring significant life sciences data. To the maximum extent possible, the vehicles and systems used in the LTBP will be the same design as those taken to Mars. This practice will allow for the operational evaluation of systems and crew performance with a high degree of fidelity (85:34-37).

1.7.7 No Other Planet or Martian Moon Investigated. Inclusion of one or both of the Martian moons, Phobos or Deimos, to the precursor missions for landing man on Mars has already been ruled out of consideration prior to the beginning of our efforts. The Program Executive Officer's decision was to concentrate our efforts on life support self-sufficiency in the Martian atmosphere before investigating the use of Deimos and Phobos. Due to the harsh penalty paid for each extra pound of weight added to the mission, the decision was made to forego visiting the Martian moons until a more robust human presence and operating capability in the Martian environment has been established. The immediacy of establishing man's permanent presence and attendant life support systems on Mars was given priority over dividing our mission resources to conduct exploratory visits of the moons looking for additional mineral resources.

1.7.8 Critical Technologies Commercially Available. Some currently immature technologies are considered essential in our mission planning. Identification of these technologies and their application to our program are covered in their respective sections of this report.

1.8 Project Team Organization and Responsibilities

The Phase II Project Ares design team is organized along project lines (reference Appendices A and B). The Project Manager is assisted by a three-man Project Planning and Integration Division whose responsibilities include scoping and integrating the overall Project Ares effort, identifying strategic goals and objectives, establishing milestones and schedules, conducting interim program reviews with the Program Executive Officer, and producing the final documentation. The other two divisions are the Transportation and Mission Divisions (reference Figure 3).

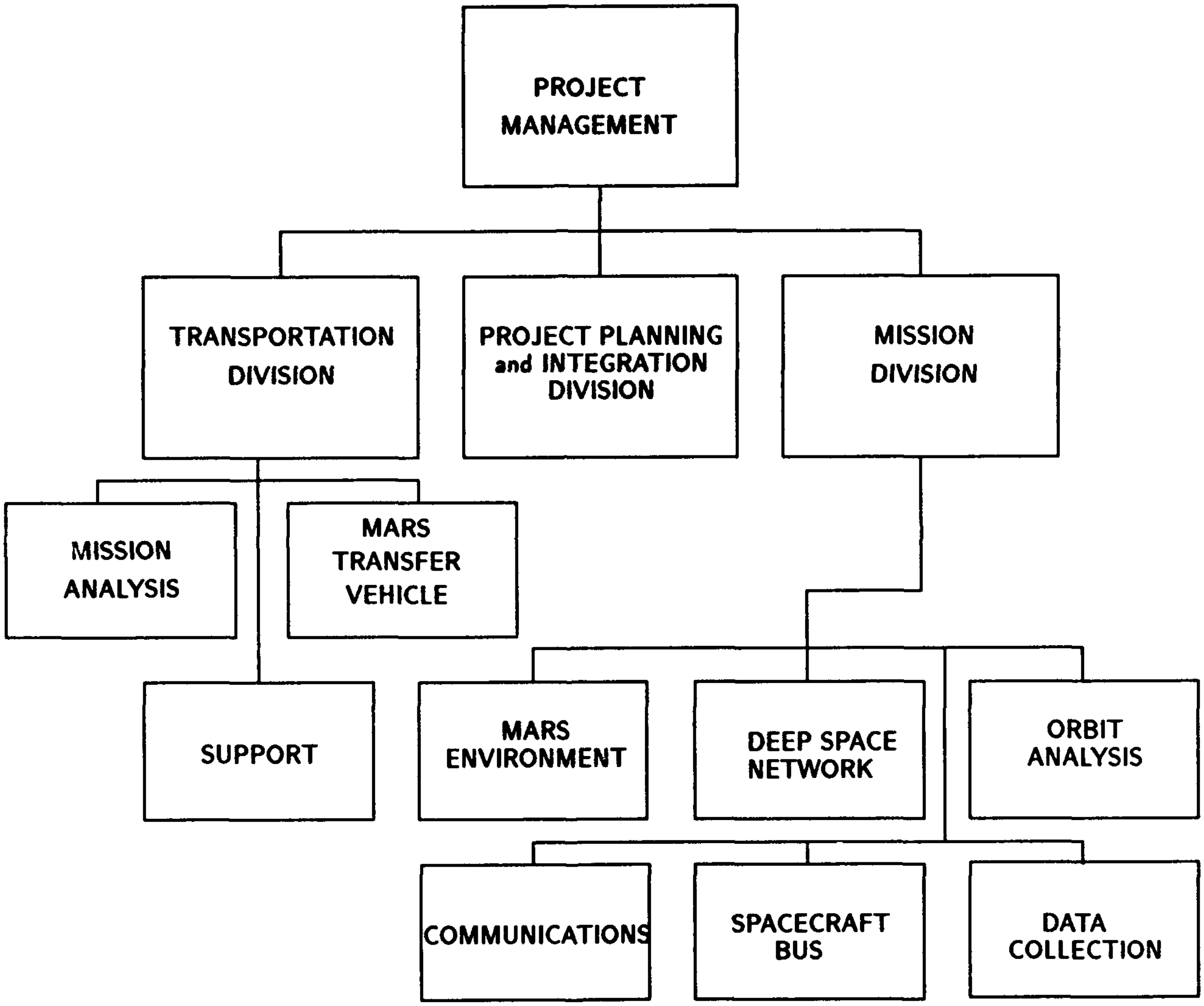

PROJECT MANAGEMENT
TRANSPORTATION DIVISION
PROJECT PLANNING and INTEGRATION DIVISION
MISSION DIVISION
MISSION ANALYSIS
MARS TRANSFER VEHICLE
SUPPORT
MARS ENVIRONMENT
DEEP SPACE NETWORK
ORBIT ANALYSIS
COMMUNICATIONS
SPACECRAFT BUS
DATA COLLECTION

Figure 3. Mission to Mars Project Team Organization.

Personnel within the Transportation Division are organized into three separate branches of responsibility: Support, Mission Analysis, and Mars Transfer Vehicle. The Support Branch will identify the launch assets necessary to lift mission payloads into Earth orbit. The Mission Analysis personnel will examine the astrodynamics options involved in an Earth-Mars trajectory. The Mars Transfer Vehicle staff will identify the subsystem requirements of the MTV, specifically examining five distinct subsystems: propulsion; power sources; navigation, guidance, and attitude control; telemetry, tracking, command, and communications; and structure and payload.

The Mission Division divided its efforts into two distinct phases, each with its own organizational structure. During the first phase, Mission Division personnel organized to research existing data on the Martian environment from Earth-based observations and the NASA Mariner and the Viking interplanetary probes; research NASA's deep space communications network capabilities and future upgrades; and research the astrodynamics involved in orbiting Mars. The second phase concerned developing the plans and requirements necessary to accomplish the overall mission. In this phase, the Mission Division organized to research three areas: Spacecraft Bus, Communications, and Data Collection. Specific areas explored were overall spacecraft design, power generation and distribution, attitude control systems (ACS), structure and thermal components, communications, TT&C, data and payload storage, and antenna configuration.

The three-division project organizational structure was chosen after recognizing that our task fit the classical definition of a project: developing a complex system within a specified period of time with both resource limitations and performance parameters. The two major advantages of the project approach we sought to realize were the designation and use of a project manager to provide a focal point for leading, monitoring, and distributing the work to be accomplished and the development of requirements and specifications for numerous mission components and hardware systems in *parallel* efforts with various groups of personnel and space expertise (66:10-11).

1.9 Summary of Findings and Recommendations

The systems engineering approach is an iterative process which ultimately leads to a final solution consisting of a convolution of compromises made along the way. This statement certainly exemplifies our efforts, especially in light of the tradeoffs made between mission requirements and transportation capabilities as a result of a preponderance of physical and technological constraints. The critical variables in this project were often complex and interdependent. Design considerations of one component most always affected a multitude of others. This fact is specifically evidenced within the detailed analysis chapters of this report. The following top-level summary is provided to highlight only the most significant decisions concluded by our research.

The Transportation Division arrived at the following conclusions:

- *Astrodynamics.* The astrodynamics of the planet's alignment dictated the feasible launch windows. When coupled with the need to minimize exposure to galactic radiation (by minimizing the transit time) the optimum launch window was established for the first manned voyage—September 2014. With this date as a benchmark, we were able to backout feasible launch windows for Phase II.
- *Exposure.* The desire to minimize the length of human exposure to radiation and other adverse space-related conditions became the primary consideration in the selection of a propulsion system. Since Phase II is unmanned, a minimum transit time restriction was not directly applicable; however, the MTV is envisioned to be the same vehicle utilized in the manned phases of Project Ares. We were, therefore, constrained to select a propulsion system that would

meet our requirements for *fast* transit times. We ultimately selected a nuclear thermal propulsion system after eliminating chemical, nuclear electric, and solar electric systems. The nuclear thermal propulsion system resulted in the best combination of thrust, efficiency, mass, transit times, and potential for growth.

- *Trajectory.* A comparison of the mass requirements for different types of trajectories made it readily apparent that a low-energy conjunction class trajectory was the optimal choice for Phase II missions.

The Mission Division arrived at the following conclusions:

- *Communications Satellites.* We envision placing two communications satellites in Mars synchronous orbit at a radius of 20,424.67 km and zero degrees of inclination. The satellites will be placed 170 degrees apart to provide maximum coverage. The Ka-band was chosen for high-data-rate links, and the C-band was selected for uplinks and communications with surface units.
- *Orbiting Surface Mapper.* A surface mapper will be placed into a near-polar orbit at 92.8 degrees. The mapper will operate in two modes—high and medium resolutions. The high resolution mode will be used to map 12 prospective landing sites at a resolution of one meter. The secondary mission of the mapper will be to map the entire Martian surface at the medium resolution. The mapper has a design life of five years, and its sensor will use silicon CCDs following a *pushbroom* motion to collect the data. High resolution data rates are calculated to be on the order of 300 Mbps.
- *Probes.* We will deorbit a total of 12 probes, packaged in fixed aeroshells, which can be delivered on a single bus. The bus will have preset codes to insert the probes over 12 candidate sites. The individual aeroshells provide initial deceleration with subsequent braking supplied by parachutes, retro-propulsion, and crushable coverings. Each probe will be powered by a lightweight radioisotope

thermal generator with a required lifetime of one Martian year. The probes will conduct atmospheric and soil analysis for six months to aid in the site selection process for our two mobile laboratories.

- *Mobile Laboratories.* Based on the information from the probes, two mobile laboratories will be deorbited to the two most promising landing sites. Each laboratory will consist of a landing and support platform (lander) and a mobile laboratory vehicle (rover). The rover-lander pair will weigh just over 1,000 kg in orbit and 750 kg on the Martian surface. The rover will be a wheeled vehicle two meters long and one meter wide. The lander is a 2.5-by-2.5 meter square with four landing legs. Both the rover and the lander require less than 100 W of electrical power each. The rover lifetime will be one (Earth) year; whereas, the lander will last 10 years.

Throughout this effort, we performed an examination of Project Ares Phase II mission and system requirements. We have highlighted many of the complexities and interrelationships that exist in a project of this scope. Although our research and computations were tedious and thorough, this report represents only a first order analysis—based on our understanding of current and foreseeable evolutions in technology. Logically, a great deal of in-depth research is required in every area exposed in this report as it relates to all phases of Project Ares; consequently, we recommend that further study be devoted to 1) developing an analogous analysis of Phases III-V requirements, 2) conducting an in-depth analysis of both identifed as well as omitted complement systems for Phase II, and finally 3) revising this paper as new technologies mature.

II. Mission Operations Environment

As a precursor to the hardware requirements and design, the Mission Division researched three key areas: the Deep Space Network (DSN); the interplanetary environment; and Mar's orbital and surface conditions. Logically, an understanding of current and future DSN capabilities to support long term, deep space projects is required before designing systems that must rely solely on the DSN. Equally as important are the conditions under which equipment must operate. Two distinctly different operating environments are apparent; deep space and the Martian arena of operations. The effects of the former is key in determining the viability of long duration transport time and hardware packaging. The latter heavily dictates equipment requirements and mission objectives.

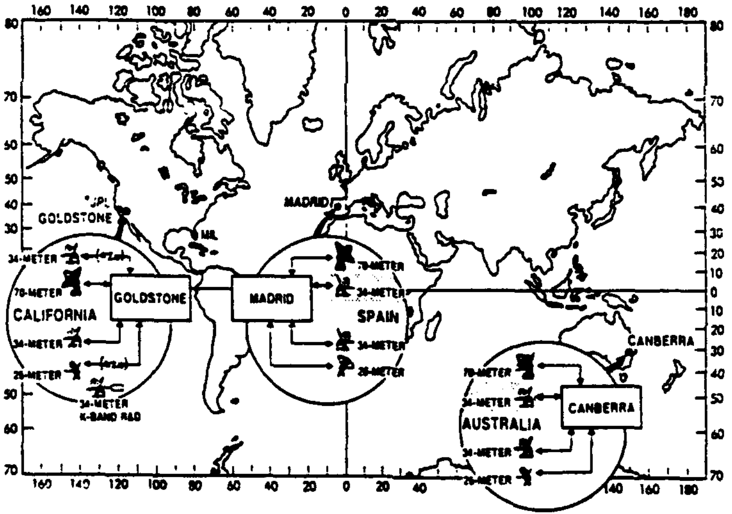
2.1 Deep Space Network

For over 30 years, NASA's Deep Space Network (DSN) has been the communication and navigation link for US lunar and deep space missions. Its primary mission is to support the operation of both manned and unmanned missions and to provide instrumentation for radio and radar astronomy in the exploration of the solar system and the universe (49:1). Overseen by the NASA Office of Space Operations and operated by the Jet Propulsion Laboratory (JPL), in Pasadena, California, the DSN is a ground-based, precision science instrument which has set high standards for remote measurement, control, navigation, and associated data-processing.

The cornerstone of NASA's planetary science programs, the DSN resources are valued at over $1.5 billion (33:1). This system supports deep space, lunar, and high-Earth orbit missions sponsored by the US and international cooperatives. The accuracy, reliability, and versatility of the network is considered by NASA to be an extension of the precise performance of the spaceborne scientific packages it supports.

The DSN was first used to support the Echo project, a 1960 experiment to use a passive satellite to transmit voice communications coast-to-coast (49:7). As we contemplate our interplanetary missions, it is evident that we will have to rely on our improved DSN capabilities to support our complex Mars operations.

2.1.1 Current Capabilities–DSN Today. The three Deep Space Communication Complexes (DSCCs) are the communication antennas and signal processing centers of the DSN. The DSCCs are located at: Goldstone, in southern California; Robledo de Chavela, near Madrid, Spain; and Tidbinbilla Nature Reserve, near Canberra, Australia (45:1). Spaced at approximately 120-degree intervals around the globe, they can provide nearly constant coverage for interplanetary spacecraft (reference Figure 4).


*JPL
GOLDSTONE
MIL
MADRID
34-METER
70-METER
CALIFORNIA
GOLDSTONE
34-METER
26-METER
34-METER
K-BAND R&D
MADRID
70-METER
34-METER
SPAIN
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26-METER
CANBERRA
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AUSTRALIA
CANBERRA
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26-METER
160 140 120 100 80 60 40 20 0 20 40 60 80 100 120 140 160 180
80 70 60 50 40 30 20 10 0 10 20 30 40 50 60 70

☐ PLANNED 34-METER REPLACEMENT WITH HIGH EFFICIENCY ANTENNA

*JPL INCLUDES THE NETWORK OPERATIONS CONTROL CENTER AND OTHER SUPPORT FACILITIES

COMMUNICATION BETWEEN JPL AND GLOBAL SUPPORT FACILITIES IS PROVIDED BY THE NASA COMMUNICATIONS NETWORK

Figure 4. DSN Global Configuration (33:248).

This support comes primarily in the form of receiving telemetry signals from the spacecraft, transmitting commands to control various vehicle operations, and generating radio navigation data used to locate and guide the spacecraft to their destinations. Secondary functions of the DSCCs include flight radio-science, radio and radar astronomy, very-long-baseline interferometry (VLBI), precise measurements of minute Earth movements (geodynamics), and participation in the NASA Search for Extraterrestrial Intelligence (SETI). The DSN is considered a world-leader in each of these areas (49:1).

Each DSCC is composed of four operational Deep Space Stations (DSSs): one is equipped with a 70-meter antenna, two are equipped with 34-meter antennas, and one is equipped with a 26-meter antenna.

2.1.1.1 Spacecraft Communication. Deep space communications are not limited by bandwidth considerations, as they are at near-Earth distances, but by signal-to-noise ratio (SNR) (33:249). The SNR limits the data rates which can be produced by any given telecommunication technology over interplanetary distances. In addition, the detrimental effect of a low SNR is increased by distance[1] and limitations on spacecraft antenna size and transmitter power. Conversely, the detrimental effects of a poor SNR are decreased by the use of a higher frequency and low-noise ground receiver (39:270).

Currently, all DSSs are equipped with X-band transmitters and receivers, the highest frequency signal handling equipment in use by spacecraft presently supported by the DSN. Of those vehicles, Magellan has the highest data rate, with a downlink of approximately 500 Kbps (63:122).

While the smaller 26-meter antenna size can support high-Earth orbits, its size is too limited to overcome the SNR problem and collect enough useful data from interplanetary missions. The 70-meter antennas however, are 7.25 times more powerful and sensitive than the 26-meter antennas and can maintain communications with spacecraft to the edge of the solar system (49:8). The 70-meter antenna at

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[1] SNR decreases with distance squared.

Goldstone was used successfully during Voyager 2's encounter with the planet Neptune in 1989, forming a communication link with a one-way distance of 4.5×10^9 km (49:8).

The 34-meter antennas are capable of supporting interplanetary-scale telecommunications at lower data rates when used alone, or at higher data rates when multiple antennas are used simultaneously via a technique called *arraying.* In this technique, the signals from several geographically separated antennas are combined in such a way as to form an effective antenna aperture much larger than any of the member antennas. This array of antennas improves the effective SNR of the spacecraft's data signal, and supports higher data rates from longer distances (63:109). Arraying techniques can also employ the DSN's 70-meter antennas.

As an example of the 34-meter antenna's solo abilities, a high-efficiency version was used to communicate with Voyager 2 at a distance of 3×10^9 km from Earth (as it passed Uranus)(49:8). When the same-class antenna was used in an array, it supported Voyager 2's Neptune encounter at a significantly increased distance and at the higher data rates required. This interagency, cooperative effort combined DSN resources with other radio astronomy facilities around the world to create effective antenna apertures large enough to capture the quality and quantity of imaging data previously thought impossible from the outer edges of the solar system (18:91).

2.1.1.2 Ground Communications. The three DSCCs communicate with the Network Operations Control Center at JPL, the DSN control node, via ground and space links through the Ground Communications Facility (GCF) co-located at JPL. The GCF provides the formatting, recording, processing, monitoring, and delivery of digital data not only within the DSN, but to external interfaces as well.

The data flow must be reliable, yet support the high data rates of the spacecraft. Computers at the GCF perform data exchanges with each of the DSCCs.

Current data throughput capacity is 280 Kbps from the DSS to JPL, with a data quality standard of 99.95 percent (45:3).

2.1.2 The DSN Tomorrow

2.1.2.1 Current Upgrades. Antennas were added and enlarged around the DSN in preparation for the Voyager 2 Neptune encounter (18:92); however, an additional 34-meter antenna is currently under construction at the Goldstone complex (49:8). The GCF Upgrade Task is a six-year project scheduled for completion in 1990. This upgrade will increase the data throughput capacity from the DSCCs to JPL by an order of magnitude to 2.27 Mbps and from JPL to the DSCCs to 224 Kbps (45:1). Mass data storage is being upgraded from magnetic tape to optical storage systems, with large gains in capacity, accessibility, and reliability. Data transfer to and from storage will then be possible at extremely high data rates: 3 Mbps per volume, multiplied by the number of volumes required. In addition, expert systems will help oversee new computing, error-correction, and networking operations (45:1–2).

2.1.2.2 DSN Evolution. Since its inception 30 years ago, the DSN and the spacecraft it supports have evolved significantly. Telemetry systems have improved by a factor of 10^{10} and navigation systems by 10^{6} (33:249). These improvements were driven by the technological requirements of the missions supported. As NASA looks into the next century, it is framing the DSN's goals in the projected requirements of SEI-type missions. Apollo-like missions to Mars and unmanned planetary explorations with launch rates varying from one every two years to one every five years are already under consideration (33:249). The DSN also recognizes the possible need for telecommunication relays in orbit about SEI-target planets.

DSN personnel have identified the following specific systems growth areas for the twenty-first century (33:250):

- Robust mission operations monitor and control.

- Network loading, coverage alternatives, and availability.
- Continuous coverage of orbiters and landers.
- Implications of *human-rated* links.
- Data standards for multi-mission and cross-support compatibility.

They also have listed the technical performance drivers as (33:250):

- Downlink data rate and quality.
- More accurate navigation and radio science.
- Error-free, gap-free information transmission to users.

In the next twenty years, DSN managers foresee the need for 10–100 Mbps data link performance, a 1000-fold increase from present rates, over trunks carrying video, multi-spectrum scanner, and synthetic aperture radar images from planetary missions. This increase will have to be within the constraints of 10^{-9} error rates, a 1000-fold decrease from current performance, for highly compressed telemetry channels and reliable remote computer downloading (33:250). Higher frequencies (32 GHz microwave and 0.5 μm optical), larger spacecraft antennas, ground arrays, and other technologies are needed to support these communication needs (reference figure 5). In addition, spacecraft storage capacities of 10^{12} bits are expected to help mitigate coverage constraints and terrestrial weather effects.

In the area of navigation, the increased accuracies needed for aerocapture maneuvers and reliance on precision orbital rendezvous and landings will require accuracies in parts in 10^{8}, an order of magnitude tighter than is currently possible. This fact, combined with 10-fold improvement in target planet ephemeris, could result in larger payload capacity due to propulsion mass savings (33:251). Performance goals for spacecraft state vectors include 5×10^{-9} radian angular measurements, ranging accuracies to 10 cm, and time correlation to 1×10^{-8} seconds (33:251). The exact

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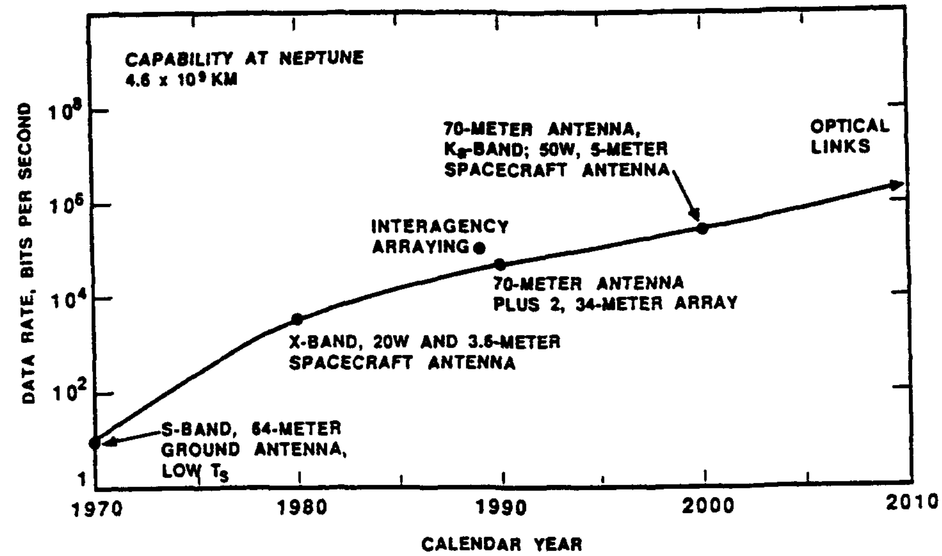

Figure 5. DSN Mission Telemetry: Support and Expectations Through 2010 (33:251).

calibration of media-induced errors and the use of intercontinental-baseline VLBI are considered critical to achieving these state vector accuracies.

For improvements in DSN *availability*, *operability*, and *reliability*, the network is looking to increased use of automation, experiential databases, expert systems, and fail-soft[2] configurations and designs (33:252–253). Automatic communication relay and navigation networks around Mars and the Moon are under consideration, as is an extension of the DSN into Earth orbit.

2.1.3 DSN Support Evaluation. The DSN is well-equipped to support Mars operations even though communications are at distances up to 1,000 times that of Earth-Moon links and the SNR ratios are one million times smaller (85:80). Certainly, such distances are small, and the signals are powerful compared to those of current missions to the outer planets. The communications aspect of navigation will be proportionally less difficult, due to the shorter distances as well.

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[2] Graceful degradation.

Data rates and throughput required for Mars operations will far surpass those currently supported for any other deep space mission, complicating the task of providing support. NASA has already recognized the most promising technologies and equipment to support missions to Mars and is incorporating them into the DSN planning processes.

NASA maintains that the DSN must continue to (33:249):

- Provide reliable high performance coverage.
- Be able to adapt to evolving objectives of mission sets.
- Maintain on-going development to provide low-risk technology when needed.
- Provide all of this at an affordable cost.

These goals emphasize that there is no better choice than the DSN to provide the quantity and quality of ground support necessary when the US goes to Mars.

2.2 Environment

2.2.1 Deep Space. Ionizing radiation represents the greatest environmental impediment to the success of this interplanetary mission. The flux particles from the Sun present the greatest problem for mission hardware during its journey from Earth to Mars. While the steady flow of particles is well characterized, major solar events such as solar flares (lasting hours to days at a time) can increase the risk of induced electronic error by three or four orders of magnitude over the normal solar event background. The probability of a major solar event occurring during a mission of this length is on the order of 20 percent[3], a significant chance considering the possible consequences of mission failure (83:1436). The three primary effects of radiation on the interplanetary spacecraft are disruption of electronic equipment due to single particle impacts, differential electrical charging effects, and degradation of materials due to chronic radiation. The second greatest risk in deep space is a collision with a micrometeor.

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[3] There has been about one such event per 11 year solar cycle since monitoring began.

2.2.1.1 Single Event Upsets (SEUs). SEUs of electronic components present a significant problem. Caused by the passage of high energy radiation through microminiature circuitry, these events cannot be totally prevented. Symptoms include sudden increases in device power consumption, garbled processor memory, and erratic control system behavior. While these occurrences should not be life-threatening to a properly designed spacecraft, continuous monitoring of spacecraft systems is recommended due to the random and erratic symptoms these errors can cause.

2.2.1.2 Differential Charging. The effects of sudden normalization of differential electrical charges among spacecraft components can also be erratic in nature. Factors contributing to differential charging include radiation particle flux density, vehicle illumination, and material and physical characteristics of the vehicle surfaces (72:1032). Bad data transmission among subsystems, *spikes* on power distribution circuits, and electrical insulator breakdown are some of the indications of unintended discharge.

2.2.1.3 Prolonged Exposure. The slow breakdown and decomposition of materials exposed to ionizing radiation over long periods represents a life-limiting factor for some spacecraft designs. The primary effect is to degrade the efficiency of exposed sensor elements and photovoltaic cells (solar cells). Eventually these components break down and can no longer support the mission.

2.2.1.4 Micrometeors. It is estimated that meteoroids have an average velocity of 11.4 km/s and a maximum of approximately 26.5 m/s (13:96). A chance collision with microscopic, high speed interplanetary particles would jeopardize the mission.

2.2.2 Martian.

2.2.2.1 Martian Space. The Martian system, composed of the main planet and two very small moons, orbits approximately 1.52 AU [4] from the Sun. The solar constant at this distance is 43.1 percent that of Earth or 590 W/m^2 on average[5]. The Martian system takes 687 days to make a complete revolution about the Sun and travels in a slightly elliptical orbit.[6] Because of this eccentricity, the Martian system's distance from the Sun varies as much as 20 percent at its the two extremes (perihelion and aphelion). In relationship to Earth's orbital plane, the Martian orbit is inclined by 1.9 degrees. Mars rotates once every 24 hours 39 minutes 35.3 seconds, making a Martian day essentially equivalent to an Earth day. Due to the difference in orbital periods, the two planets pass one another once every 26 months. Combined with the slightly eccentric nature of the Martian orbit, the distance between the planets as they pass one another varies over a 15 year cycle (synodic period).

The two moons of Mars, Deimos and Phobos, circle the planet in opposing directions. Deimos, the smaller of the two, occupies a supersynchronous orbit at an altitude of 20.12×10^3 km. Having a mass of 1.83×10^{15} kg. Deimos has a mass ratio of 2.9×10^{-9} with Mars.[7] Phobos rotates counter to the spin of Mars and orbits at an altitude of 6.0×10^3 km. It has a mass of 2.72×10^{16} kg and a mass ratio of 4.2×10^{-8} to that of Mars. Due to the relatively small size of these moons, neither is expected to have significant effects on satellite orbits.

2.2.2.2 Martian Atmosphere. As measured by the Soviet Mars spacecraft, the structure of the Martian magnetosphere is comparable to Earth's. Evidently due to the smaller and cooler molten core, the intensity of the Martian magnetosphere is approximately 0.0003 that of Earth (32:49). This will affect any plan to use a Martian magnetic field to manipulate satellite orbits and will preclude its use for surface navigation. The weak magnetic field, in turn, causes a reduced capability to capture and hold charged particles in the upper atmosphere, creating a weak ionosphere at best.

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[4] 1 AU = 149.5×10^6 Km, the approximate distance from the Earth to the Sun

[5] With a high of 708 W/m^2 at perihelion and a low of 488 W/m^2 at aphelion. Reference Section 3.8.2.4, paragraph five.

[6] The Martian orbit has an eccentricity of 0.1.

[7] Mars has a mass of 6.42×10^{23} kg or 1/10 that of Earth. Martian gravity is, however, 1/3 that of Earth.

Data obtained from Mariner and Viking prove that the Martian upper atmosphere consists largely of CO_2, with small amounts of N_2, Ar, CO, O_2, O, and NO and shows strong mixing to heights in excess of 120 km. Carbon monoxide and Nitrous Oxide are enriched here in relation to their presence in the lower atmosphere. The ionosphere shows a single F1 layer with a maximum at 130 km. Ion number densities measured at this peak are on the order of 100,000 per cubic centimeter, with 0_2^+ comprising 90 percent of the layer and CO_2^+, the remaining 10 percent. The plasma frequency (f_{min}) for this layer, which sets a lower limit for ground-to-orbit communications is calculated by;

$$f_{min} = \sqrt{\frac{Ne \times (1.6022 \times 10^{-19})^2}{(8.8542 \times 10^{-12}) \times (9.1094 \times 10^{-31})}} \tag{1}$$

For the electron densities, Ne, observed, fmin is calculated to be 17.84 Mhz, which corresponds to a maximum passable wavelength of 16.82 meters. The lower atmosphere consists of CO_2 (95.32 percent), N_2 (2.7 percent), Ar (1.6 percent), O_2 (0.13 percent), CO (0.07 percent), and trace amounts of Ne, Kr, Xe, and O_3, as well as variable trace amounts of water vapor. From winter to summer, the amount of water vapor over the polar caps increases 85 ppm. This increase strongly suggests a permanent cap of frozen water is present.

The water vapor in the atmosphere is concentrated near the surface and is supposedly transported between hemispheres in the changing seasons. Precipitation on any appreciable level can be ruled out. Peak water vapor concentrations are on the order of 100 precipitable microns.[8] A thin haze is postulated at dawn, dissipating by noon. This, with the addition of other atmospheric aerosols, could complicate data interpretation for the morning hours.

Martian atmospheric pressure varies seasonally by about 30 percent. Viking measurements ranged from 6.8 to 8.4 mbar. It is believed that this gradient is seen over the global scale. Winds will be much stronger than observed if this range of pressures was confined to a local scale. Winds are mild, generally less than 20 m/s, with highly repetitious daily patterns during the summer months. These patterns, generated by the global circulation, are modified by local terrain.

Temperature at the Viking landing sites ranged from 150 Kelvin (K) at night to 240 K in the midafternoon. The orbiter's thermal mapper showed ranges of 130 K to 290 K in the band from 30 degrees north latitude to 60 degrees south latitude. Viking data supports the theory that CO_2 condensation occurs locally near the equator before dawn.[9] Historically, large dust events occur in the southern hemisphere, from spring to summer, when the planet is near perihelion and surface temperatures are conducive to strong dynamic activity.

Planetary albedo was found to vary from 0.089 to 0.429, with a mean of 0.214 and standard deviation of 0.063. Viking data showed a marked absorption minimum at wave number 7231.58 per centimeter, which corresponds to the spectral line for silicon. The corresponding wavelength, 1.3828 μm would be optimal for any infrared (IR) imaging of the surface, which the landers determined to be roughly 44 percent silicate by weight. The bandwidth of this window is roughly 8 angstroms, and would also encompass the spectral lines for chromium (7230.05) and manganese (7230.5), which were not detected in the Viking soil samples.

The only atmospheric obstructions in the visible wavelengths are clouds composed of dust, frozen water, and/or CO_2 ice. The ice clouds detected by Viking were

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[8] Typical Earth values are on the order of precipitable inches.

[9] CO_2 condensation temperature: 149 K

41

not noticeable in corresponding IR scans due to their small optical depths in the IR. The few discrete dust clouds imaged in the optical were too small to have thermal signatures.

2.2.3 Summary. The environment in deep space and at the Martian system provides a great number of challenges to Project Ares. Most significantly the interplanetary journey will expose delicate equipment and human tissue to high levels of radiation as well as high energy particles. The systems we design must be capable of long term exposure in deep space without significant degradation to mission equipment from radiation and particles. More important to the operational structure of the project is the capabilities of the DSN to support high volume, high speed communications. NASA is responding to the needs of the SEI by planning for DSN upgrades in the form of better ground networks, ground terminals, and space based relayed systems. To support future manned missions to the Martian surface, more must be known about Mars' weather, chemical composition, and geologic structure[10].

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[10] We acknowledge the proper prefix is *areo* for Mars, but for clarity of understanding, we will use *geo* throughout the report.

III. Mission Systems

The objective of this phase of Project Ares is to develop an in-depth understanding of the Martian theater of operations. The data we gather will form the basis for critical decisions regarding succeeding missions. This chapter addresses the specification of mission systems which perform data collection and relay. Due to the complicated nature of the assigned project, only a few systems of the mission have been fully explored.

We begin by identifying mission objectives and requirements, after which we will discuss the mission profile as part of the concept of operations. This mission profile forms the cornerstone of specific hardware requirements and design specifications and is therefore given special attention. Here we also enumerate our high level assumptions to provide a basis for technological expansion and system design. The bulk of this chapter deals with the analysis of individual systems that will be used to support our concept of operations. Each of the succeeding subsections provides for system objectives, assumptions, and design analysis.

The subsections are formed along functional boundaries. They deal with communications, data collection, and spacecraft design. All three areas involve multiple vehicle consideration and provide for an integrated design approach. The chapter is concluded with a summary of design decisions and recommendations.

3.1 Objectives and Requirements

3.1.1 Objectives. The task of lifting man into near Earth orbit and then flinging him tenuously across 50 million kilometers of empty space to a barren planet, hostile to life as we know it, is a dangerous and costly endeavor. Every effort must be made to reduce the risks involved in such an endeavor. The primary purpose of this phase of Project Ares is to gather data pertinent to the sustainment of human life and mission hardware. Having examined existing data on Mars and the associated environments, we plan to support the above purpose with the following objectives:

- *Solar Effects.* The solar effects during an interplanetary trip between Earth and Mars are not well defined. More accurate and detailed data is required on the effects of radiation, microgravity, and galactic rays on human tissue. Further study of solar flares outside Earth's magnetopause is of prime importance. Our understanding of solar flares directly relates to our ability to sustain human life while in transit and while on Mars.
- *Uncover Hazards.* The ability to sustain human life on Mars will be adversely influenced by two types of hazards: 1)the corrosive characteristics of the Martian atmosphere and dust storms and 2) the more direct risk of organic contamination of life support equipment. The former hazards are identified and quantified through research. The latter category is much more difficult to itemize due to our limited experience with universal organic forms. In most cases, we simply do not know the questions, let alone the answers. Therefore, we must be able to totally insolate ourselves from the atmosphere and surface in the early manned-phases to perform scientific study of the environment. The requirement to insolate ourselves directly impacts Phase II of Project Ares. We must provide data on the first category of hazards such that non-corrosive materials can be applied to system designs used in latter phases of the project.
- *Resources Availability.* Current data is insufficient to build resource generation hardware for life support systems in the Martian environment. It is critical we gather detailed information concerning chemical constituents of the Martian soil. Water, a prime element to human life, is theorized to exist on Mars but has not been found in sufficient quantities to support life. Further examination of surface and subsurface soil is required at various locations on Mars. Rock composition analysis has not been conducted to date and will be a top pri-

ority in this phase of the project. Core samples must be acquired to provide information on subsurface minerals, ores, and toxicity.

- *Weather.* From Viking data we have a good idea of the yearly Martian conditions on the weather front. Weather however, has a tendency to cycle over decades, centuries, and even millennia. All surface probes must collect and relay weather data.
- *Global Information.* Data on volcanic, tectonic, and magnetic activity is needed on a global scale. Inferences about the planet's core can be made based on this information and has yet to be collected comprehensively.
- *Site Selection.* Phase II culminates with the site selection for future manned missions.

3.1.2 Requirements. Secondary to reducing mission risk is testing of hardware concepts and creating a remote infrastructure that can support a variety of future scientific and life support roles. As the infrastructure grows, so too will its reliance on Earth based resources. Cargo supply ships will have to be sent regularly. Communications systems will become the only respite to the inhabitants as well as the only means of capitalizing on hard earned research. It is natural that each system we produce and send to the far away outpost be enduring and flexible. A slow process of building systems that increase capabilities and yet remain compatible with existing hardware is a second means of reducing overall project cost. From this second purpose that we form our set of mission requirements:

- *System Reliability.* Should a support system, such as a communications satellite, fail to deliver its full operational capability as a component of the overall Mars infrastructure, future mission equipment designed to utilize that system will be severely effected. Certainly the replacement equipment will be delayed and possibly even canceled given the large deployment costs and timelines involved in replicating the support system capability. Therefore, system reliability must be comparable to man-rated systems. Project timelines must include replacement and redundant capabilities to preclude single-point loss of mission capability.

- *Hardware Robustness.* Systems produced in the early phases must form the basis for future constellations of equipment. For example, based on the project timeline spanning 20 years or more, communication relays will have to maintain operations for 8 to 12 years initially. Surface probes that provide navigational beacons must provide enduring radio frequency (RF) signals for 10 to 15 years. Surface reconnaissance from orbit, however, is not required to support future phases of the project and can have a relaxed lifetime requirement of five operational years. Likewise, ground mobile laboratories are of limited capability and can have relatively short life expectancies (6 to 12 months).
- *Interdependence of Operations.* The systems we design must conform to the overall program architecture. Components have to be designed that do not interfere with one another and provide for future expansion. As an example, communication systems should allow for time and frequency multiplexing of data while leaving room for future systems to make use of unutilized bandwidths.
- *Contingency Operations.* Even though the spacecraft systems are highly reliable and robust, the capability to destroy sensitive equipment abounds in space, which means that we can never plan for every possible situation. Therefore, the concept of operations must include contingency plans in case a mission critical failure occurs. With contingency plans in place before hardware is built, system designers can provide equipment capabilities to perform these operations even if the baseline design has no other justification for the capability. An example of a contingency capability would be to allow the orbiting reconnais-

sance vehicle to send data directly to Earth in the event of a communications relay anomaly.

- *Site Selection.* Because the introduction of man onto Mars will undoubtably upset the local environment, the site must be chosen so that man's life support systems can be reasonably isolated from the surrounding Martian environment. A second consideration for site selection will be the number of scientifically interesting locations within a reasonable distance from the site.

3.2 Mission Profile and Assumptions

3.2.1 Mission Profile. Based on the Phase II objectives and requirements, it is best to develop a mission profile which will form the basis for system designs. A solid mission profile not only aids system designers, but also provides for a basic understanding of the types of activities and capabilities being developed. Understandably, the profile will change according to innovations in technology, but for the most part, these will be minor in a project of this magnitude.

The profile we worked from involved two Mars excursions separated by approximately 52 months.[1] The first trip to be launched in 2001 would include two systems; two communication relays and an orbiting, surface mapper. The second trip, to be launched in 2005 would ferry 12 surface probes and two mobile laboratories to Mars. The original concept was to take only one communications satellite in the first stage of this phase and augment this constellation with a second vehicle in the following stage. Due to Mars Transfer Vehicle (MTV) weight limitations we normalized mission payloads between the two trips. For this reason, the second communications satellite was shifted to the first trip with a resulting mass balance of approximately 6.5 metric tons. The basic description of each stage's payload follows.

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[1] This corresponds to two conjunctive periods. Further discussion is provided in chapter 4.

- *Communication Relays* – These are robust communication relays that will provide a central command, control, and communications node for the near Martian theater. The communications satellite as envisioned will provide trunk[2] reception and transmission with Earth based DSN. The satellite is to provide for bulk data storage, compression, and transmission selection based on ground Earth-commands. The communications relay must provide for data and command transmission to the mapper and eventually to Mars-based probes. Communication cross-link capabilities for expansion of the relay satellite network must also be present.
- *Martian Surface Mapper (MSM).* This vehicle's main mission is to map the Martian surface to a predefined resolution. The spacecraft must house several other upper-atmospheric experiments, and a store and dump capability used to transfer surface hardware data to the relay vehicle. The objective of the mapper will be to discern specific locations within predefined candidate landing regions (100 Km^2) as landing sites for stationary surface platforms. The locations will be chosen based on Viking surface data.
- *Environmental Probes.* The 12 probes will be carried in an autonomous dispenser which will orbit the planet and be deorbited for surface evaluation atmospheric constituents, climatology, seismology, surface reconnaissance, and volcanic activity. The probes will have a common mission package augmented by specific instruments tailored to the local conditions expected at the landing sites (e.g. photon flux at poles and equator). Some probes may have cameras capable of transmitting still imagery at low data rates. A primitive navigational beacon will also be integrated into each probe.
- *Mobile Laboratories*[3]. The laboratories will accompany the environmental probes but remain in orbit a maximum of six months prior to deorbit. Once

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[2] A trunk contains many data streams at a high data rate.

[3] The mobile laboratories are later referred to as the rover or rover-lander.

primary and alternate sites are selected from the 12 candidate locations, the labs will be deorbited near the appropriate probes and operate in a mean radius of 31 km of the beacon. The labs will perform predefined experiments in the vicinity of the beacon, such as core samples analyses.

3.3 Standard Mars Orbits

Standard orbits were calculated and evaluated for two types of satellites: communications and mapping. The communications satellite orbit was chosen to optimize communication and telemetry relay between communications equipment on the Martian surface and Earth, and between the mapper and Earth. The mapper orbit was designed for an optical mapper with consideration given to resolution and ground swath requirements.

3.3.1 Communication Satellite. The orbit chosen for the communications satellite(s) is a Mars-stationary orbit[4]. A stationary position relative to the surface would provide constant line-of-sight (LOS) between the communications satellite and the surface sites within the communications satellite's field-of-view (FOV). The relatively high altitude of a Mars-stationary orbit combined with the 25.19 degree inclination of Mars to its solar orbit would provide less eclipsing between the satellite and Earth than a satellite in a low, equatorial orbit. An equatorial orbit was more desirable than an inclined orbit since the Mars surface is initially going to be explored near the equatorial regions, and an equatorial orbit would provide more LOS time than an inclined orbit. A Mars-stationary communications satellite would also provide a more predictable target for the mapping satellite for uplinking of data.

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[4] The correct term is aerostationary. For ease of understanding, we have used the term Mars-stationary.

For a circular orbit (eccentricity, $e = 0$), the radius of orbit is equal to the length of the semi-major axis, a, in the following equation:

$$TP = \frac{2\pi}{\sqrt{\mu}} a^{3/2} \tag{2}$$

Rearranging equation 2 and solving for a (where μ, the gravitational parameter for Mars is 42828.287 km^3/sec^2, and the desired time period, TP, is the length of the Martian sidereal day (88642.663 sec)) yields an initial orbital radius of 20,427.67 km. The oblateness of Mars, however, affects the mean motion of the satellite resulting in an orbit with a faster than desired revolution rate. Correcting for the oblateness of Mars, an orbital radius of 20,428.78 km is obtained. This would put the satellite at an average altitude of 17,034.84 km above the Martian equator.[5] The ideal communication setup would be three Mars-stationary communications satellites spaced 120 degrees apart providing uninterrupted[6] communication relay between Martian surface equipment (at other than the extreme polar regions), the mapping satellite, and the communications satellites.

A constellation of two communications satellites would provide constant communication coverage,[7] excluding two regions of the equator that would not be in direct LOS with either of the two communications satellites. If the communications satellites were directly opposed to each other, the two *cones* with no coverage would have a half-angle of approximately 9.6 degrees measured from the center of Mars, with a resultant total length along the equator of 1,133 km. These two lengths would mean that approximately 10.6 percent of the Martian equatorial region would not have direct contact with either communications satellite. This deficiency could be alleviated by relaying through the mapper which would have LOS contact with any point on the Martian surface at least once per day.

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[5] **Average Martian equatorial radius is 3,393.94 km.**

[6] **Apart from the approximately two weeks caused by solar conjunction.**

[7] ibid

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A single Mars-stationary communications satellite would only have contact with less than 40 percent of the Martian surface. If there are no landers on the surface, then this is a moot point. A single communications satellite, though, could provide direct LOS contact 50 percent of the time with a satellite in an orbit as low as 48 km above the Martian surface. For any satellite link expected to last more than several hours, the orbit will be much higher; and the higher the satellite, the longer (percentage wise) that satellite will be in contact with the communications satellite.

Mars is not perfectly circular at the equator, but can be modeled by two ellipses with a common minor axis.[8] The major axes of the two ellipses have nearly the same length, $a = 3,394.67$ km and 3,393.21 km. The spheroid formed by these ellipses (ignoring topography) nearly coincides with the average surface form (35:23). Since the planet has no bodies of water, the density of the planet, appears to be relatively constant radially and thus, is expected to cause very few perturbations to a Mars-stationary satellite.

3.3.2 Mapping Satellite. The initial purpose of the mapping satellite will be to chart a chosen band of latitudes centered about the equator. The satellite orbit must be designed to take into account the optical characteristics of the mapper, atmospheric effects, possible perturbations caused by Mars, communications and datalink with the communications satellites, and possible future taskings of the mapper satellite.

A circular retrograde orbit was chosen in order to keep the satellite's altitude above Mars as constant as possible. This is desirable for an optical system so that focusing adjustments are minimized.

As low an orbit as possible is required in order to achieve the desired resolution. This consideration must be weighed against the effects of atmospheric drag at lower altitudes which would cause the orbit to decay. In the case of Mars, "the limit of the real atmosphere ... is shown to be ... about 130 km" (35:75). Thus, initially an altitude of 150 km above mean equatorial radius was chosen to begin calculations for the mapping satellite's orbit. Even though the atmosphere is nearly non-existent at this Martian altitude (much thinner than Earth's), some drag will occur and must be taken into account when determining maneuvering fuel reserves. An increase to a 360 km orbit would essentially eliminate drag, decreasing the number of orbits per Martian sidereal day from 13.8 to 12.7. The longer lifetime of the satellite more than offsets the slight increase in time to map the surface.

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[8] The north/south minor axis with $b = 3,376.78$ km.

The satellite's inclination to the Martian equatorial plane was determined after consideration of primary mission, perturbations, and future taskings. A 90 degree will provide complete mapping of the entire surface; however, the initial mission only calls for mapping of a desired zone about the equator. It has also been calculated that a 90 degree inclination is one of 11 critical inclinations that would cause a satellite's orbit to wander in inclination and eccentricity (46:70). A low inclination orbit of less than 45 degrees would provide thorough mapping of only the near equatorial regions, as desired, but would prohibit (through large expenditure of fuel) future use of the same satellite for use in mapping latitudes from 45 degrees to 90 degrees. The lower inclinations are also more affected by precession of orbit due to the oblateness of Mars. Even though J_2 is known for Mars, the calculated precession may not be as close to the actual precession as desired which would result in extra expenditure of maneuvering fuel to correct for desired ground track. A higher inclination orbit has less precession and would thus lessen the impact of any errors between calculated and actual precession.

As the mapper satellite is primarily an optical gatherer, it is desirable to have as much light as possible reflected back to the satellite. This favors a sun-synchronous orbit with a track overhead at noon. A report on the Mars Observer Camera, however, recommended that the orbit view each portion of the planet at 1400 hours local time giving satellite images very similar to the successful National Oceanographic and Atmospheric Association Television and Infrared Observational satellites (NOAA TIROS) used on Earth space(29:8). Determination of the inclination of the orbit was an iterative process involving the effects of Mars' oblateness upon the mean motion of the satellite:

$$\bar{n} = n_o \cdot \left[1 + \frac{3 \cdot J_2 \cdot a_e^2}{2a_o^2\,(1 - e_o)^{3/2}} \times \left(1 - \frac{3}{2}\sin^2(i_o)\right)\right] \tag{3}$$

and precession of the line of nodes of the satellite:

$$\Omega = \Omega_o - \frac{3 \cdot J_2 \cdot a_e^2 \cos(i_o)}{2a_o^2\,(1 - e_o^2)^2}\bar{n}\,(t - t_o) \tag{4}$$

where: $\bar{n}$ = corrected mean motion

n_o = mean motion at epoch

J_2 = first order secular variation for Mars (0.001964)

a_e = radius of Mars at equator (3393.94 km)

a_o = major axis of satellite orbit

e_o = initial eccentricity of satellite orbit

i_o = initial inclination of satellite orbit

Ω = precession of line of nodes in time $t - t_o$

Ω_o = position of line of nodes at epoch

t_o = epoch

To obtain a sun-synchronous orbit, the precession of the line of nodes must match the mean motion of Mars about the Sun (1,886.519 arcsec per Martian sidereal day). Thus at a 360 km altitude with a zero eccentricity orbit, the desired inclination was calculated to be 92.7 degrees. The 360 km altitude orbit gives a period of 6,991 sec with a precession of 0.524 degrees per Martian sidereal day. The resultant surface ground speed is 3.05 km/sec.

3.3.3 Probes and Rover-Landers. Orbits for the spacecraft carting the surface probes and rovers were evaluated based upon the profile presented earlier in this chapter. The 12 probes are to be deposited at various pre-determined sites selected using information obtained by the mapper. A logical orbit to use for the probe canister would be the same as that used by the mapper. The 92.7 degree inclination and 360 km altitude orbit would provide near global coverage except directly over the poles. The inclination and orbit altitude provide for 12.68 revolutions per Martian sidereal day, with ground traces that would cover the sunlit side of the equator (ascending node) at 230 km spacings over a period of seven days. Thus, if the probes could be maneuvered by up to 115 km either side of their track during the landing sequence, all the probes could be landed at the desired surface locations within the first seven days of orbit. The mapper, if still operational at that time, could be used to confirm the probe landings, since it is in the same orbit.

Since 12 landings in seven days will be a very demanding sequence and we will need to confirm the serviceability of each probe after it has landed, the landing sequence will probably be stretched out over a longer time period. The chosen orbit will be beneficial in this regard, since the ground traces will essentially repeat every eighth day.[9] Another advantage of this orbit is that it is sun-synchronous, and the probes would be landing during daylight hours.

Since the surface probes and rovers would be transported from Earth to Mars on the same run, it is desirable to place the rover-landers in the same inclination as that of the probe spacecraft. Thus, the MTV would not be required to make any energy intensive out-of-plane maneuvers. Because the rovers can not remain in orbit for six months (following probe deployment), a higher, more stable orbit is desired to conserve station keeping fuel. However, a higher orbit requires additional reentry fuel. We must balance the need for stability against the additional fuel requirements. Data gathered on mapper orbital parameters will lend a great deal of information on fuel requirements of a 360 km circular orbit. The rover-lander pairs will have to be inserted into a somewhat stable orbit where, without the expenditure of station keeping fuel, the canister would be in no danger of entering the atmosphere (130 km) after six months, or more if potential mission delays ensue. The projected parking orbit is at 97.2 degrees and at an altitude of between 360 and 1000 km.

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[9] Actual repeat trace is 21.3 km west of first trace.

3.3.4 Moons. The Martian moons were evaluated for perturbations upon satellite orbits. Deimos will come closest to any of the proposed satellites, coming within 300 km at its closest approach to the communications satellites. With a mass ratio of 2.9×10^{-9} compared to Mars, its effects upon any of the satellites would be small but not negligible. Phobos is slightly larger, with a mass ratio of 4.2×10^{-8} to that of Mars, but its closest approach to the mapper satellite is about 6,000 km. Thus, neither of the Martian moons is expected to have any significant impacts upon the proposed satellite orbits.

There was some interest in placing a communications platform on or around the moons of Mars. After examining this possibility, we determined that putting a communication station on any of the moons would restrict the communications FOV. Communications with Earth will be possible only about half the time, since the side of the moons facing Mars[10] (necessary to communicate with units on the Martian surface) would only be visible by Earth through roughly half of the moon's orbit, not including the eclipsing by Mars itself. Thus a communications setup on either of the two moons is not practical. Putting a satellite in orbit about any of the moons was as impractical as placing one on the moon due to the effect of Mars on the satellite's motion. In time, the satellite would either escape from the moon or crash into it (19:49-51).

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[10] Both moon rotation rates are of the same period as their orbital period–the same side always faces Mars, much as the Earth's moon does with Earth..

3.4 Communication, Command, and Control

3.4.1 Concept of Operations. How any mission to Mars handles its communication of data is a vital measure of its chance of success. If it fails to communicate outgoing task instructions and returning observations both quickly and carefully, it will directly impact the effectiveness of Mars mission hardware by reducing the effective resolution, coverage, accuracy, and perhaps, lifetime of mission packages. This task of data communication across millions of kilometers of deep space is complicated by extremely fast data rate requirements, near-perfect data quality standards, complex networks of spacecraft and probes within the operating theater, and vast separation in distance and time between the operators and the machines they need to control. Thus, this Mars mission will require a communication system which is faster, more exact, and more autonomous than any known space system.

Using technologies which are in the planning stages today makes such a system highly feasible. Their analysis and assumptions are discussed in following sections, but the segments of the overall system are shown in Figure 6 and listed below:

- *Earth Support Stations.* This capability will be provided by an enhanced DSN.
- *Command Uplinks.* All instructions are sent directly to the communications satellite. Mapping satellite and surface unit commanding will also route through the communications satellite.
- *Data Trunks.* These tracks will return spacecraft and mission data to Earth via the communications satellite and space-based DSN antennas.
- *Mars Relay and Communication Satellites (MaRCoS I and II).* MaRCoS I and II will store and relay all data returning from the Mars operation, and relay commands to the mapping satellite and surface rovers. Both communications satellites will be launched with the first Phase II launches and placed approximately 170 degrees apart in synchronous orbit, far enough apart to preclude

only a 19-degree longitudinal sector of the surface from coverage,[11] yet still permitting line-of-sight contact for the communications satellite crosslink, as described in Section 3.4.3.3. MaRCoS II however, will be placed in storage mode until two to four years after arrival on station.

- *Mars Surface Mapper (MSM) satellite.* The MSM stores and regularly transfers large amounts of data to the prime MaRCoS for relay to Earth.
- *Command/Data Broadcast.* Instructions for, and data from, Mars surface units will be routed through the MaRCoS spacecraft.

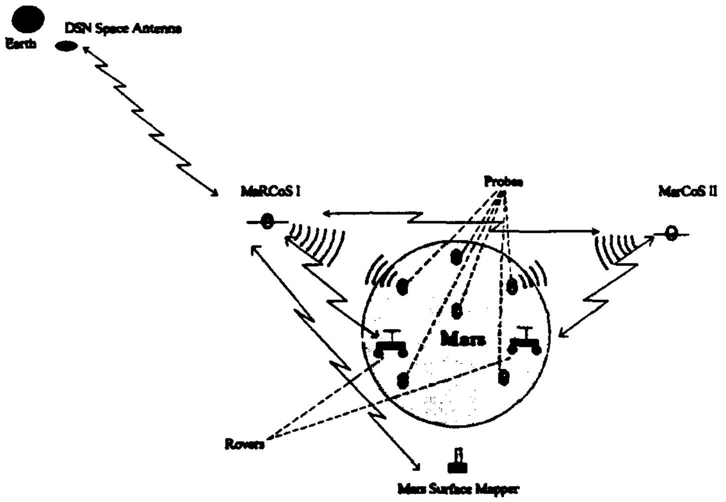

Figure 6. Overview of Mars Communication System.

Not only is such a system capable of supporting Phase II operations on Mars, but it is a necessary stepping stone to proving the necessary technology for the follow-on human presence. However, the important capabilities of Mars surface-to-surface communication and the Earth-Mars-Earth exchange of high-rate video necessary to support manned missions will not be demonstrated by the Phase II communication system. Instead, these capabilities could be demonstrated separately during the dress-rehearsals on the Moon.

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[11] This is referred to as the Blind-spot.

3.4.2 Overall Objectives. The primary goals for a Mars communication system revolve around its ability to properly handle the vast amounts of data involved. The system should be based on the most reliable, fastest data links available. Certainly, the 100–600 Kbps links used by current deep space systems would be very constraining on this project, which involves thousands of megabit-size images of the Martian surface and countless scientific measurements coming simultaneously from 12 surface probes and two surface rovers. In addition, the faster the data is transferred, the easier it can be lost, so the system must work around Earth's weather interference, severely limit noise-induced link errors, and avoid data loss due to buffer overflow at communication nodes. The Mars communication system should deliver the data in near real time.

Once on station in 2002, the communication satellite should have a useful lifespan of 10 years, a figure supported by current lifetimes of similar Defense Satellite Communication System, Phase III (DSCS III) spacecraft. This, combined with the two to four year storage of MaRCoS II, will permit a sufficient overlap with next generation systems launched in preparation for the phases involving human presence slated to begin in 2014. All system components and segments should have built in redundancy, eliminating potential single-point failures wherever possible. MaRCoS II will provide important redundancy for surface unit data relays, as well as a proof-of-concept in use of a satellite cross-link to work around occultations[12] during Martian equinoxes. In addition, it can be brought out of storage to take over primary communication relay duties in the event of the catastrophic failure of MaRCoS I.

3.4.3 Specific Requirements, Assumptions, and Decisions

3.4.3.1 Earth Support Segment. Given the mission and past performance of the DSN, we expect this network will grow to provide all the data handling for the envisioned Mars communication system. Specifically, we assume that the DSN will be upgraded from its current configuration, discussed in Chapter II, to include space-based antennas, functioning similarly to the current Tracking and Data Relay Satellite System (TDRSS). Such platforms are currently under consideration and will serve three primary purposes. First, they would convert between the Mars-pointing Ka-band (32 GHz) link, which offers a higher data rate, and the Earth-pointing X-band (8.4 GHz) link, the highest frequency band to easily penetrate the atmosphere (96:3). Second, the extension of the DSN into space would increase network reliability by shunting downlinks to alternate ground terminals when weather or equipment failures threaten data losses (33:253). Third, it avoids the costly addition of another 70-meter antenna at each Deep Space Communication Complexes (DSCC) by dividing the large time demands placed on the Mars-facing DSCC among the remaining complexes as needed. It also permits the use of 34-meter antennas to receive the downlinks from geosynchronous orbit. All this supports the concurrent need for the DSN to continue supporting other deep-space programs requiring the same antenna resources required by the Mars mission.

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[12] The state of being hidden from view.

DSN antennas in Earth orbit must be capable of buffering data as required by a drop in data rate between the downshifting of frequencies during the downlink to Earth. Optical storage devices similar to those later described on the communications satellite could be used. Further design specifications are not detailed in this report.

Ground-based antenna arrays, with their large resource overhead, will not be used regularly to support Mars operations. Also, existing optical storage systems will be enhanced even further to warehouse all data from the communications satellite and more emphasis placed on expert systems to track, store, and route to user agencies the terabits of new data from the Martian theater.

The DSN will also support normal spacecraft commanding functions associated with Mars mission spacecraft via C-band (5 GHz) uplinks multiplexed together on the ground and then relayed from ground antennas via the MaRCoS spacecraft. C-band has performed well as a command link during past deep space missions and offers the added benefits of requiring low transmit power and all-weather usefulness (10:4). Phase II Mars operations require no new technologies in this area.

3.4.3.2 Earth-Mars-Earth Communication Trunk. This trunk will consist of an extremely high frequency return leg towards Earth and a lower frequency leg carrying vehicle instructions away from Earth. It will be maintained between Earth and the primary MaRCoS spacecraft. For the mission data returning from the Mars mission, a one-way Ka-band data trunk from the communications satellite will be used. Ka-band technology is currently breadboard-validated (33:276) and should be ready to deliver the performance required. This band is also within the range allotted by the DSN to deep space exploration (10:4).

The Ka-band can support data rates of 10 Mbps at Mars distances (39:272). Two fully redundant trunks and related hardware will ensure that a single-point internal hardware failure will not terminate the mission. These two downlinks will be separated sufficiently in frequency to avoid self-interference and will be subdivided into multiple channels which can be rapidly reconfigured to carry whatever blend of image, scientific, and state-of-health data the users and operators require. The extremely high Ka-band frequencies minimize signal-to-noise problems and provide maximum bandwidth, making the very high data rates achievable (23:398). They also require less transmit power onboard the spacecraft (10:4). While optical communications have even higher projected data rates (100–300 Mbps) and show much promise for future applications, the technology is currently only conceptual and will not be sufficiently mature to support an operational mission in the 2001 timeframe (33:276). However, optical communication test packages could be included on one or both communications satellites, within the overall 3,000-watt communication system power constraint, should breakthroughs occur in sufficient time for proper integration.

A single uplink trunk to the prime MaRCoS will simplify DSN resource usage by not requiring separate antennas. Since no manned or telerobotic operations are planned prior to the waning years of the communications satellite's lifetimes, we assume that uplink data rates on the order of 10 Kbps should suffice for command and database uploads. Current DSCS-III satellites, with reprogrammable random access memory (RAM), use only 1 Kbps uplinks. The absence of real-time video uplink requirements permits both this lower data rate and the use of a lower frequency communication link. All this is compatible with the use of C-band uplinks, each consisting of eight data channels of medium bandwidth, one primary and one backup for each of the two communications satellites and the mapping satellite, as well as two to share between the two surface rover-lander pairs. The surface probes will be non-commandable.

The Earth-Mars link will be interrupted by periods of occultation. One cause for such an interruption is when Mars is between the communications satellite and Earth. This occurs during Martian equinox, with the longest occultation projected to last 1.31 hours (10:6). Once MaRCoS II is activated, the satellite crosslink will provide a *workaround* for such link interruptions. Communication will also be blocked for up to 17 days every 2.3 years when the sun is between Mars and Earth[13] (10:6). A satellite crosslink will have no affect in this case. These occultations drive the requirement that mission controllers be able to send commands for immediate storage and delayed execution. This ability to hold and later execute such commands will be resident in each affected spacecraft and ground unit.

3.4.3.3 Mars Communications Satellites. Much like any other communications satellite, MaRCoS I and II will serve as relays, but they also must meet the new technological challenges posed by the use of Ka-band and operations millions of kilometers from Earth. These challenges include exact pointing control of the antennas, extremely large storage capacities, and unattended data handling.

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[13] This configuration is referred to as solar conjunction.

There will be three Ka-band antennas onboard the communications satellite: an Earth-transmit antenna, a mapper-link antenna, and a communications satellite-crosslink antenna. The Earth-transmit antenna will be an 8-meter, parabolic, focal-fed reflector. This antenna size is larger than that currently used in deep space, but is similar to those already under design (39:276). The size is dictated by signal strength and background noise parameters. It will be positioned so that it may slew as needed to ensure near-continuous[14] Mars-Earth communications. Such slews should be small since the satellite's attitude control system will always maintain a fixed orientation with respect to Earth. The mapper-link antenna will be a 1-meter, high-gain, diplexed dish placed on a boom and fully gimbaled to permit it to track the mapping satellite throughout its visible orbit. A duplicate antenna will permit the cross-link channels between communications satellites. This data crosslink between MaRCoS I and II will be similar to those already planned for the TDRSS follow-on program (26:1-1). Both the mapper-link and cross-link antennas will pass commands in one direction and data in the other.

One potential drawback of using extremely high frequencies are the very tight pointing constraints placed on the transmitting antennas. The impact of these constraints are magnified many times at interplanetary distances. However, antenna pointing accuracies of 130 nanoradians are projected by 1998, which will allow effective Ka-band communications to Earth and within the Martian theater (56:30). We assume that this same technology will enhance the accurate, rapid slewing of the mapper-link antenna required by the high relative velocity between MaRCos I and the mapper. Similar operations are currently planned for the TDRSS follow-on spacecraft at similar, but even higher frequencies. In both Earth-MaRCoS and MaRCoS-mapper links, the initial antenna positioning will be via a command-stored program. Pointing will then be maintained by monopulse autotrack algorithms (26:1-23).

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[14] Except during solar conjunctions.

The MaRCoS spacecraft will also be equipped with five C-band antennas. The C-band command uplink from Earth will access the satellite via two hemispheric, receive-only antennas mounted on opposite faces of the spacecraft and providing a near-spherical reception pattern (8:3-47). This pattern will be important in the event the satellite begins to tumble and ground controllers are attempting to regain attitude control. The command uplink will be demultiplexed onboard the prime MaRCoS and instructions for other units broken out by their identifying frequency and packet headers before being routed to the proper destination. After activation of MaRCoS II, this will include relaying commands to the second communications satellite, as well. Dual, movable, 3-meter Mars-pointing antennas on each MaRCoS will each then transmit a single C-band spot beam to the location of each surface rover-lander pair within view. The beam will provide a larger footprint than the rover's 31 km operating radius. The 3-meter dishes onboard MaRCoS should also be fully diplexed and used to receive the uplinked surface data from the probes, landers, and rovers. Onboard processing will slew the MaRCoS antennas to optimize the signal strengths of the surface uplinks during normal operations. Command-stored programs will initially acquire the rover-lander pairs prior to descent to the Martian surface.

Three redundant strings of traveling wave tube amplifiers (TWTAs) and their corresponding pre-amplifiers and filters will be allocated to both the Earth and Mars links. TWTAs are expected to remain the choice for large RF output, linear amplification well into the twenty-first century (4:829) and triple redundancy increases reliability in this critical node. Only one of each type of amplifier string will in use during normal operations. The Mars-pointing string will operate in the C-band region and reamplify uplinked rover commands before relaying them to the surface. It will also continuously output a beacon for rovers on the surface to use in autotracking the communications satellites and in navigation. The Earth link must output 70 watts of RF power in the Ka band (39:272) and, assuming antenna size limited efficiency, will draw 500 watts of electrical power from the spacecraft bus. The required C-band RF output of the Mars link will be significantly less due to the smaller distances. Also, little or no interference is expected from the Martian atmosphere, based on the Martian environment evaluation. A similar design proposal for a Mars communications satellite, using a very high gain antenna design and small focused beams, required only 75 watts of electrical power for a focal feed array to produce a spot-beam of 30 watts RF power towards the Mars surface (10:5). Each MaRCoS requires only two spot beams, providing coverage for up to two surface rovers within a single communications satellite's FOV.

The data storage capacity onboard a MaRCoS is integral to the success of its relay mission. Its primary function will be to avoid data losses due to both the limited capacity of the data links and interruptions of those links during occultations. The communications satellite will use an array of ten memory devices similar to those already under development. The specific justifications for this very large storage capacity are provided in the following section. The NASA spaceborne optical disk recorder development program has identified a system utilizing two 14-inch disks in a *jukebox-type* design (80:272). The performance goals of this system, as of 1988, are:

- 120 Gigabit capacity (9.6×10^{11} bits).
- 1.8 Gigabits per second transfer rate (1.8×10^{9}).
- Concurrent input/output operations.
- Reconfiguring architecture.
- 300 Mbps transfer rate.
- 10^{-10} corrected bit error rate.

We estimate power requirements for such a system at 1,000 watts, given current system specifications and some increase in efficiency.

This system will combine a 10:1 data compression capability with the 10-unit array of improved storage devices to allow the fully robust communications satellite to continually store a large amount of Mars data even during the 17-day solar conjunction. Multiple units provide a fail-soft capability to increase reliability. MaRCoS I and II will normally operate in a continuous first-data-stored, first-data-transmitted downlink while data is resident onboard, and will continue to store whatever data is uplinked from the Mars theater while it has memory available. However, ground commands can be used to alter the order of data blocks in the queue. Due to the limits of the storage capacity, the potential exists for some data to still be lost due to the memory becoming loaded to capacity. These same ground commands will then be used to choose which data is most beneficial. The optical disk controller will have a self-test capability to identify degradation and reconfigure the memory to maximize capacity, automatically or by ground command.

The combination of huge quantities of data, link occultations, and light-time link delays[15] drives the need both for powerful means of routing commands intended for multiple destinations and the data arriving from multiple sources. Automatic switching, path selection, and buffering are required. A reliable expert system onboard the communications satellite must be capable of autonomously directing data handling during normal and contingency operations and during periods of link occultation. Such a system is central to structuring, storing, managing, and moving data within the system (39:275). For example, the amount of data from each source placed in the Earth-bound data trunk will be determined by an expert system accessing reprogrammable software which will alter multiplexer and channel assignments. To facilitate these autonomous decisions, command sequences and data sources will be identified by channel frequencies and packet headers containing source, destination, and priority information.

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[15] Up to 20 minutes in one direction between Earth and Mars.

3.4.3.4 The Mars Surface Mapper Satellite. The MSM will map an area of 2.98 km by 0.5 meters every 1.61×10^{-4} seconds. Data gathered over this period is referred to as a *picture* of data. The high resolution (HIRES) mode is designed to generate 6,000×8 bits at 1.0 m resolution thus, the MSM will gather 300 Mbits of data for each second over a possible landing site. This equates to 6,211 pictures per second. The mapper, traveling at 3.05 km/s,[16] will cover a potential landing site in about 3 seconds. At HIRES, a total of 900 Mbits will be generated per orbit. Assuming worst-case of two sites encountered per orbit, a maximum of 1.8 Gbits per orbit could be generated. For medium resolution (MEDRES), the satellite will continually map the entire planet, so it will generate 0.67 Mbits of data every second. For one two-hour orbit, this equates to 4.8 Gbits of data. Combined, the mapper will generate 6.6 Gbits of data per orbit. Allowing for 25 percent more data for the handshaking protocol necessary for transmission to the communications satellite, the total amount of data transmitted to the communications satellite is 8.25 Gbits. The mapper will be in line-of-sight of the communications satellite for approximately 3,000 seconds per orbit. At a data rate of 10 Mbps, the MSM can transmit up to 30 Gbits of data per orbit a factor of three over what is actually required.

Due to the large amounts of data generated by the MSM and surface units, the success of the mapper is contingent upon high performance (high rate/high capacity) memory systems. The mapper will have a storage system identical to the communications satellite: 1 terabit capacity (10^{12} bits) with a bit transfer rate capability (300 Mbps) that exceeds the bit rate of the communications system. This storage capability will allow storage of over 100 orbits of MEDRES mission data before the memory is filled. The MSM storage units will also have the same automatic functions as the communications satellite, such as the self-test and reconfiguration functions, as well as the ability to alter the order of data block transmission.

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[16] Ground Speed

The mapper will utilize two antenna types. First, data and telemetry will be transmitted to the communications satellite by a fully gimbaled, high-gain dish antenna, via two fully redundant Ka-band channels. This 1-meter, transmit-only antenna will be located on a boom away from the mapper body, permitting it to physically track the prime MaRCoS as it passes overhead using the same methods as the communications satellite antenna tracking units. Two fully redundant strings of TWTAs with corresponding pre-amps and filters will be allocated to each channel, with the one operational string drawing 150–200 watts of power. This electrical power requirement is based on a similar low Earth orbit (LEO) system proposed for NASA (26:1-7). In the event of a failure aboard the communications satellite, the antenna will be slewed toward Earth and a significantly lower data rate telemetry and data stream invoked; a technique used by NASA with interplanetary probes.

The second type of antenna will be a hemispheric C-band antenna. The MSM will use two of these, each located on opposite faces of the mapper. These antennas will provide contingency operations should anomalies preclude use of nominal command paths. If the communications satellite is disabled, commands will be received directly from ground controllers via these antennas, while limited telemetry and data will be transmitted via the Ka-band antenna as mentioned above. If the mapper begins tumbling, the near-spherical coverage provided by the hemispheric C-band antennas will allow ground controllers to implement contingency commands. This type of antenna configuration is currently used on the DSCS III satellite program. Again, two fully redundant C-band channels will be used, each with two fully redundant hardware strings.

3.4.3.5 Surface Command/Data Relay. Since probes, landers, and rovers will not be placed on the surface of Mars until the second launch of Phase II, the

MaRCoS I and II satellites will both be on station and ready to support their operations. Probes will be small and limited in power budget and number of measurements. A further limitation will be the need to transmit their data via omnidirectional antennas to MaRCos I or II, 17,000 km overhead. All this demands that the data rate be very low, less than 100 bps, over C-band. Each probe will use a solid-state transmitter and will have its own narrow frequency assignment to easily identify the data source. Probes will not be commandable. The 19-degree blind-spot due to the 170-degree separation of MaRCoS stations will be placed where no probes are located by adjusting the communications satellite stations.

Landers will carry the rovers from storage in orbit to the Martian surface. While in orbit and during descent, the landers will receive commands and transmit data via its low-gain, dipole antenna and low-power transmitter and receiver to and from the MaRCoS using C-band. However, contingencies would permit use of the high-gain antenna onboard the rover to relay data and commands directly to and from Earth. After they have arrived on the surface, the landers will continue transmitting data to its assigned MaRCoS.

The rovers will be much more capable and able to support a larger power supply and high-gain antenna, thus they will use a higher data rate in the C-band. The estimate of the data output of the numerous sensors and experiments, as well as the video feeds, is 25 Kbps. This should be transmitted to the communications satellites in real time, but limited storage capability will be carried onboard in case delays become necessary. A high-gain, 0.5-meter antenna on each rover will autotrack the C-band carrier and beacon from its MaRCoS, permitting servos to continuously optimize antenna pointing. Commands will also be relayed from the communications satellite to the rover via this antenna. While uplinks will be on separate medium-band channels, primary and redundant downlink channels will be shared between the two rovers. This will be possible due to the anticipated physical separation of the rovers on the Martian surface. In a contingency, the rovers will transmit data directly to Earth at a significantly reduced data rate.

3.4.3.6 Node Failure Analysis of Mars Communication System. In order to more fully evaluate the robustness of the Mars communication system design, we have described the desired mode of operation given the following failures:

- *DSN Antenna/Ground Station.* If a DSN antenna or ground station fails, we will use alternate, visible DSS either directly in clear weather or via space antenna relay. No degradation of capability is expected.
- *Primary Earth-Mars-Earth Data Trunk.* If the primary Earth-Mars-Earth data trunk fails, we will use the redundant trunk having the same data rate with no degradation of capability.
- *Primary Communications Satellite or Mapper Command Link.* If a command link fails, we will use a backup channel having the same data rate with no degradation of capability.
- *MaRCoS.* If we lose a MaRCoS, we will use the remaining MaRCoS to relay data from the mapper and surface units positioned on the visible hemisphere. The result will be the loss of data from approximately half of the surface units.
- *Primary MaRCoS Earth-Transmit Only.* If a traveling-wave-tube amplifier (TWTA) is lost, we will use crosslink to relay data to second MaRCoS for downlink to Earth. Degradation: unable to continuously maintain Mars-Earth data trunk during Martian equinox.
- *MaRCoS Mars-Transmit Channel.* If we lose a MaRCoS Mars-transmit channel we will use an alternate channel. No degradation of capability is expected.
- *MaRCoS Mapper Data Link.* If we lose the MaRCoS mapper data link, we will use the redundant link having the same data rate and experience no degradation of capability.

- ***MaRCoS-MaRCoS Crosslink.*** **If the crosslink fails, we will initiate separate downlinks. Degradation: Loss of contact with the communications satellite during Martian occultations.**
- *Mars Surface Mapper.* If we lose a MSM, we will continue to relay surface unit data over communications satellites, but will lose all imaging data.
- *Rover Transmit Channel.* If we lose a rover transmit channel, we will use an alternate channel with no degradation of capability.

3.4.4 Summary. If critical technologies are available by the dates foreseen by researchers, a robust, fully capable Mars communication system can be ready to support Project Ares Phase II missions. An overview of the data links proposed is illustrated in Figure 7. Using Ka-band communications, optical storage, and expert systems will optimize the data-gathering capabilities of the mission equipment. Failure of these technologies to mature in the coming years will severely jeopardize the ability of MaRCoS to provide reliable, high performance connectivity within the Martian theater and between Earth and Mars.

3.5 Martian Surface Mapper Payload

3.5.1 Concept of Operations. The MSM will be sent in the second launch of Phase II. Before any probes land on the surface of Mars, the landing site must be examined in great detail and declared safe in which to land (at least from catastrophic size boulders or other geologic dangers). The primary mission of the MSM is, therefore, landing site certification. It will map in great detail 12 potential landing sites selected from existing Mariner and Viking data. The MSM has the added responsibility of mapping the entire surface at somewhat lower resolution for studies of Martian meteorologic and climatologic concerns and conducting other scientific experiments.

3.5.2 Specific Objectives, Requirements, and Assumptions

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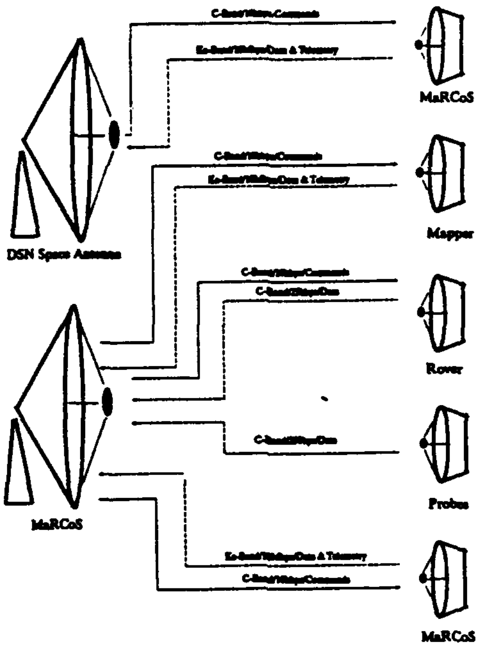

Figure 7. Communication Links.

3.5.2.1 Specific Objectives. The objectives of the MSM are as follows:

- *HIRES Mission.* To support the goal of establishing a permanent manned presence on Mars by mapping 12 potential landing sites at 1.0 m resolution .
- *MEDRES Mission.* To provide global, synoptic views of the Martian surface and atmosphere in order to study meteorological, climatological, and related surface changes during the course of the mission by mapping the surface at medium resolution.
- *SCIENCE mission.* To conduct other scientific experiments that do not have a direct impact on the primary mission, yet will increase humankind's knowledge of the universe.

3.5.2.2 Requirements. The MSM is designed to acquire images of the surface and atmosphere of Mars for qualitative and quantitative interpretation of

Martian climatology and topography(29:5). Orbiting at an altitude of 360 km, the MSM will have a ground speed of 3.05 km/sec[17]. The need to address questions dealing with Martian meteorology, climatology, and geoscience require that the data be collected over a range of time and spatial resolutions in order to examine the dependence and/or interdependence of these phenomena.

To support the HIRES mission, the MSM will map 12 potential 10 km x 10 km landing sites at 1.0 m resolution. The ability to distinguish objects which could potentially be threatening for surface probes and/or manned landers drove this degree of resolution and came about as a compromise between the more stringent requirements put forth by the Stafford Report of May 1991 and the less stringent requirements of Viking Site Selection and Certification Document of 1981.

A wide-angle camera, designed to provide a much lower resolution of 200 m, will cover a much larger area and is the sensor to support the MEDRES mission. This wide-angle camera will also be used to obtain pictures of the atmosphere and return medium resolution global maps as often as once every twenty-four hours (29:1).

The tertiary mission of the MSM is the SCIENCE mission and the mapper instrument complement is as follows (61:5):

- Pressure Modulator Infrared Radiometer (PMIRR)
- Magnetometer/Electron Reflectometer (MAG/ER)
- Thermal Emission Spectrometer (TES)
- Gamma Ray Spectrometer (GRS)
- Mars Observer Laser Altimeter (MOLA)
- Radio Science (RS)

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[17] 360 km was the maximum orbit to achieve a 1.0 m resolution with the available optics.

3.5.2.3 Assumptions. There are only a few assumptions made in this section as far as future technology is concerned. The plans for the MSM closely resemble the Mars Observer Satellite (29:1-8) which was originally scheduled to be launched in September of 1991. For the most part, the technology is mature and readily available, and will most certainly be advanced to the degree necessary for the scheduled launch date of this project. Any assumptions made as far as future technology will be pointed out in the appropriate places throughout the report.

Because the bulk of the MSM's mission will be completed within two years, it has a design life of five years. The last three years will be, for the most part, devoted entirely to MEDRES weather mapping.

3.5.3 Martian Surface Mapper Analysis.

3.5.3.1 Resolution Limitation. Before we discuss the design for the main sensor on the MSM we must know what limits our resolution.

Limits of Resolution. The ability of a lens to produce distinct images of two point objects very close together is called the resolution of the lens. The closer the two images can be and still be seen as distinct, the higher the resolution. There are two principle factors that limit the resolution of the lens. The first is lens aberrations. Because of aberrations, a point object is not a point on the image but a tiny blob. Careful design of compound lens (or use of aspherical mirrors/lens) can reduce aberrations significantly, but they cannot be eliminated entirely. The second factor that limits resolution is diffraction, which cannot be corrected for because it is a natural result of the wave nature of light.

From introductory physics, we know that because light travels as a wave, light from a point source passing through a slit is spread out into a diffraction pattern. A lens, because it has edges, acts as a slit. When a lens forms the image of a point object, the image is actually a tiny diffraction pattern. Thus, an image would be blurred even if aberrations were not present.

The best resolution possible (assuming no aberrations) is diffraction limited resolution. For a lens, or any circular hole, the image of a point object will consist of a circular central peak (called the Airy disk) surrounded by faint circular fringes. The central maximum has an angular half-width given by:

$$\theta = \frac{1.22\lambda}{D} \tag{5}$$

where λ is the wavelength of the light used and D is the diameter of the lens.

If two point objects are very close, the diffraction patterns of their images will overlap. As the objects are moved closer, a point is reached where you cannot tell if there are two overlapping images or a single image. Where this occurs could be judged differently by different observers. An accepted criterion, the Rayleigh Criterion, states that two images are just resolvable when the center of the diffraction disk of one is directly over the first diffraction minimum in the diffraction pattern of the other. Because the first minimum is at the angle $\theta = \frac{1.22\lambda}{D}$ from the central maximum, the two objects can be considered just resolvable if they are separated by this angle θ. This is the limit on resolution by diffraction due to the wave nature of light.

The original design for the MSM called for a resolution of 0.25 m. At this resolution, the diameter of the collecting optics would have to be $\approx$ 0.80 m. While this diameter optics would be easy enough to manufacture, the associated data rate would be unmanageable[18].

3.5.3.2 MSM Optical Systems. There are many different techniques which could be considered for landing cite certification, yet optical imaging is considered the most promising. Optical imaging has been used on earlier missions to Mars and on the Voyager missions to the outer planets. As a result, optical imaging has achieved a high level of technological confidence and maturity (74:592).

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[18] Please refer to Section 3.5.3.4 for calculating data rates.

The MSM uses silicon charge coupled devices (CCDs) mainly because silicon CCDs are responsive over optical wavelengths and the maturity of silicon technology has allowed manufacturing of large, linear arrays of detectors at a relatively inexpensive cost.

The detector array was chosen to be a linear *push broom* device rather than a two-dimensional framing array. Both the *push broom* and the framing array use a concept wherein the motion of the spacecraft generates the frame by exposing each line (or two-dimensional array) as the spacecraft moves along its trajectory. While the framing array offers the advantage of increased signal integration time,[19] it requires a slightly larger telescope FOV and discontinuous platform motion. It is also technically limited by the large, two-dimensional CCD arrays needed and the high data rates necessary for readout (74:592). The push broom concept uses a linear array of CCDs rather than a two-dimensional array, thereby making manufacture and readout easier. This narrow FOV system has a spectral bandpass of 500-900 nm.

Because the FOV of high resolution systems is usually quite small, the MSM is equipped with a wide angle, medium resolution, linear array correlation imager. The narrow FOV and the wide FOV systems will be used together to provide qualitative and quantitative photographic information. The medium imager provides a benchmark for the high resolution imager, thereby enabling scientists to tell exactly where the high resolution images are taken with respect to well-known Martian surface features. This wide angle system will consist of two lenses and one focal plane device. One lens will have a blue bandpass filter (400-450 nm), the other will have a red bandpass filter (575-625 nm) (29:4). Table 1 gives the specifications for our optical system and Figure 8 shows a comparison of resolutions.

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[19] The time that the individual detectors are integrating the signal from a single exposure.

Table 1. Optical System Specifications.

	Narrow Angle System	Wide Angle System
Focal length	5 m	11.3 mm
Focal stop (F-Stop)	f/10	f/6.5
Aperture	0.50 m	.0017m
Number of pixels	6000	4000
Pixel size *	6.95 μ m	6.95 μ m
Spectral bandpass	500 - 900 nm	
Blue		400 - 450 nm
Red		575 - 625 nm
Digitization	8 bits/pixel	8 bits/pixel
Resolution	1.0 m/2 pixels @ 360 km**	200 m/pixel @ 360 km
Angular FOV	8.3 mrads	2.46 rads
Ground swath @ 360 km	2.98 km x 0.50 m	320 km x 221 m

* Denotes an assumption relying on the progress of future technologies. Presently, the minimum silicon detector pixel size is approximately 10 μ m.

** Because of the difficulty in producing diffraction limited mirrors/lens, this does not represent a diffraction limited case, thereby allowing for some (though be it small) imperfections in the surface of the mirrors/lens.

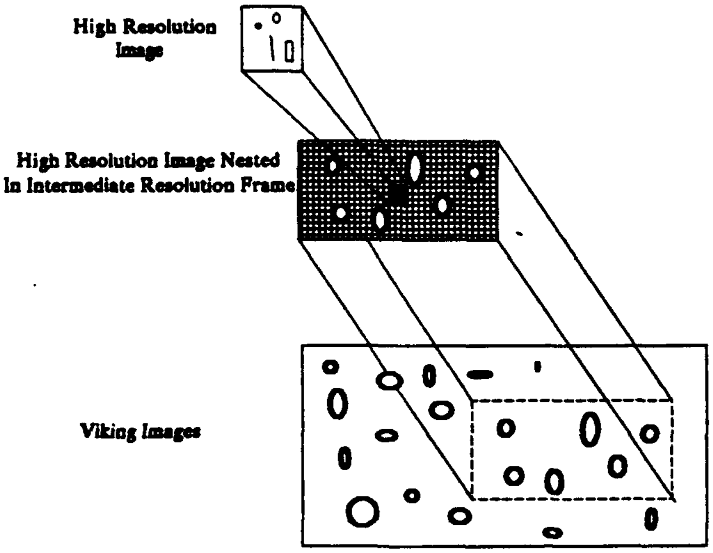
High Resolution Image
High Resolution Image Nested In Intermediate Resolution Frame
Viking Images

Figure 8. Resolution Comparison.

3.5.3.3 Tertiary Payloads. As was noted above, the MSM will conduct tertiary missions. There are six of them.

- *Pressure Modulator Infrared Radiometer (PMIRR).* The PMIRR is an instrument which will determine the temporal and spatial distribution, abundance, sources and sinks of volatile materials and dust over a seasonal cycle. Global measurements of vertical temperature and pressure profiles will allow an exploration of the structure and general circulation characteristics of the Martian surface (55:3). The expected data rate from the PMIRR is 462 bps (61:7).
- *Magnetometer/Electron Reflectometer (MAG/ER).* The MAG/ER is an instrument which will measure the magnetic field of Mars. In addition, this experiment will investigate the Mars-Solar wind interactions. The energy spectrum and angular distributions of electrons will be examined (50:2). The expected data rate from the MAG/ER is 2079 bps (61:7).
- *Thermal Emission Spectrometer (TES).* The TES is an instrument which will be used to determine the composition of surface minerals, rocks, and ices; to study the composition, particle size, and distribution of atmospheric dust; to

locate clouds and determine their height, temperature, and condensate abundance; to study the condensate properties, processes, and total energy balance of the polar ice caps; and to measure the thermophysical properties of the Marian surface material (55:4). The expected data rate from the TES is 4016 bps (61:7).

- *Gamma Ray Spectrometer (GRS).* The GRS is an instrument which will measure the elemental composition of the surface of Mars. The elements of particular interest are H, C, O, Na, Mg, Al, Si, Cl, K, Ca, Ti, Cr, Mn, Fe, Ni, Th, and Ur. Measurements will have a spatial resolution on the order of 360 km. The GRS will also detect and measure the energies of intra- and extra-galactic bursts of gamma rays and the environmental neutron flux (55:1). The expected data rate from the GRS is 2025 bps (61:7).
- *Mars Observer Laser Altimeter (MOLA).* The MOLA is an instrument which will determine the topography of Mars which will lead to a better understanding of the internal structure and global tectonics. It will also facilitate study of volcanic flow volumes and gradients, impact cratering, and the effects of topography on atmospheric circulation (55:2). The expected data rate from the MOLA is 2160 bps (61:7).
- *Radio Science (RS).* The RS is an instrument which will provide seasonal variations of the total gas content and vertical structure of the atmosphere, including the change in atmospheric opacity during dust storms (50:2). Research did not yield any expected data rates from RS.

3.5.3.4 Expected Data Rates, Handling/Storage. The optical sensors will contribute the largest share of data and will drive the requirements for the data rate considerations. In order to get an idea of the types of data rates necessary, a few calculations are needed.

First, the high resolution sensor will be addressed. Because the projection of the pixel on the ground in the along-track direction is 0.5 m and the satellite's velocity is 3.05 km/s, the sensor's maximum sample time for no-gap coverage for each high resolution exposure, t_{sh}, is:

$$t_{sh} = \frac{0.5 \text{ m}}{3050 \text{ m/s}} = 1.639 \times 10^{-4} \text{ sec} \tag{6}$$

The expected data rate from this raw data is: total bits per image per t_{sh} .

$$\text{Data Rate} = \frac{48000 \text{ bits}}{1.639 \times 10^{-4} \text{ sec}} \approx 300 \text{ Mbps} \tag{7}$$

Second, the medium resolution sensor is addressed. Because the projection of the pixel on the ground in the along-track direction is $\approx$ 221 m and again the satellite's velocity is 3.05 km/s, the sensor's maximum sample time for no-gap coverage for each medium resolution exposure, t_{sl}, is:

$$t_{sl} = \frac{221 \text{ m}}{3050 \text{ m/s}} = 7.246 \times 10^{-2} \text{ sec} \tag{8}$$

The expected data rate from this raw data is: Total bits per image per t_{sl}.

$$\text{Data rate} = \frac{48000 \text{ bits}}{7.246 \times 10^{-2} \text{ sec}} \approx .670 \text{ Mbps} \tag{9}$$

Finally, from the tertiary payloads section, we know all of the secondary payloads contribute in aggregate, approximately 11,000 bps. The total data rate for recording the data on the mapper is just the sum of the tertiary contribution, plus Equations 7 and 9. The minimum data rate for recording is then on the order of 301 Mbps.

As given in a previous section, the data rate from the MSM to MaRCoS will be on the order of 10 Mbps. The data, therefore, must be written to optical read/write disks on board the MSM. This on-board storage will be a buffer where the data must be held while the transmission to MaRCoS is taking place. In the event that line-of-sight between the MSM and MaRCoS is lost, the size of this buffer is sufficient such that the storing of five orbits worth of data is possible, under the assumption that only two potential landing sites are encountered per orbit. The data will be transmitted using a first in, first out process.

3.5.4 Summary. The MSM is a near polar orbiting surveyor with the primary mission of landing site certification. It supports this mission by imaging, in the matured optical regime, the surface with HIRES and MEDRES systems. The MSM also has the added responsibility of mapping the entire Martian surface, at somewhat lower resolution and of conducting other scientific experiments.

The MSM is an indispensable cog in the machine supporting the ultimate goal of establishing a permanent manned presence on Mars, for without the detailed mapping of the Martian surface and the insight gained to Martian meteorology/climatology, no surface probes nor manned craft could be guaranteed a suitable and safe landing site.

3.6 Martian Surface Probes

Prior to any extended manned presence on the Martian surface, a more detailed study of surface conditions, climate, geography, and geology, over a more global scale is required. The Viking data, while our most current and detailed source of knowledge, only covered two possible landing sites, merely scratching the surface. This data will also be at least three decades old before the first phase of any mission to Mars gets off the drawing board. A network of small surface probes, in conjunction with one mapper and two communications satellites provide a cost-effective method for reaching our mission objectives. The network would, by nature, also aid in site selection for future missions (21:1).

3.6.1 Objectives and Design Criteria. We need to understand the planet-wide processes along with their spatial and temporal variation. Due to the number of landing sites required, the individual probes will be kept small and simple. Limiting the measurement regime to the surface/atmosphere interface (21:1) will aid in meeting this requirement. If we assume a well mixed atmosphere, the entry data of the two Viking probes should provide a sufficient model of the middle and upper atmosphere. Because the first several manned missions will be limited to the boundary layer environment (altitudes less than 0.5 km), we must limit our investigation to this region. Later, perhaps, the first manned missions can deploy vertical wind/temperature profilers; such data will aid in any air transport of weather fronts and enhance forecasting ability. High-priority investigations that can be performed in this regime include:

- Meteorology and Climatology
- Seismology
- Geochemistry and Boundary Layer Atmospheric Constituents

The only one of the above objectives that can be performed within a short timespan is the geochemistry experiment. This is only limited by the time required to collect and analyze samples. The meteorology and climatology missions require data to be gathered over at least a Martian year, so that any seasonal cycles may be witnessed. The seismology experiment will require a long duration measurement due to the infrequency of seismic activity. This limits designs to those capable of operating in the extremes of the Martian surface environment for a period of at least Martian year. With this consideration ruled out, simple battery power in favor of solar or nuclear energy as the prime power source (21:1). Our power team, based on the need for battery storage during the Martian night, also ruled out photovoltaic power supplies. The chief concern was the need for a battery capable of surviving repeated cycling to 150 K and possibly below. Based on the availability of small light Radioisotope Thermal Generators (RTG) with adequate outputs, we have centered our designs around nuclear-thermal sources. The use of fissionable materials at the site of an eventual manned presence drives the need for controlled entry.

3.6.2 Network Configuration. Burke and Mostert recommend a network of at least 20 small probes, all capable of gathering meteorological data, with a minimum of eight capable of core sampling and at least 12 carrying seismology equipment (21:2). However, their proposed design is capable of carrying equipment for all the experiments while still keeping the craft small and easily replicated. If we choose this design, keeping in mind the Viking missions showed that the higher latitudes would be rather inhospitable to a manned presence and thus reducing our coverage of the higher latitudes, we could get by with a smaller number of probes. One concept will be to limit our observations to the 12 sites proposed by the Stafford commission.

The easiest configuration to obtain would use 20 probes equally spaced in area.[20] This configuration, however, leaves a large space between data points with large data gaps developing in the event of failure of one or more probes. If, however, we reduce the coverage of higher latitudes we can decrease the data gaps in the more hospitable latitudes. We would then have a squashed, 20-sided polyhedron configuration which should be relatively easy to obtain.

Thus, two extremes for configuration exist. At one end is the site-specific configuration using pre-determined sites, which are not necessarily equally spaced around the globe, and would require a complex insertion program. At the other end is a squashed, 20-sided polyhedron configuration with more probes equally spaced in a band around the globe, and requiring less coordination of insertion. By limiting the number of probes and preselected sites, we sacrifice the scientific usability of the network. A network of irregularly spaced meteorological stations is of little use, particularly to a forecaster on the trailing edge of a data sparse area. The opposite configuration, while more beneficial to future science, may completely miss crucial site-specific data, for example, a pocket of a hazardous elements.

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[20] Picture Mars as a 20-sided polyhedron with one probe in the center of each face.

Since our primary mission is site-selection, not scientific exploration, and since the lifetime of the instruments would make it highly unlikely that any would still be useful when the manned mission arrived, the logical choice is the site-specific configuration. We must keep in consideration the added computer memory on the entry vehicle necessary to perform the complex insertion. We should not rule out the future application of the 20-sided polyhedron network in future phases.

3.6.3 Lander Delivery Considerations. Burke and Mostert discuss a method of delivering several probes from a single spacecraft (21:2). Each probe is attached to its own aeroshell[21], making each probe capable of safe, independent entry. Several probes are attached to a bus, such as the one in Figure 9, which would hold six probes.

Our design is simpler and requires one bus capable of delivering 12 probes. This allows for one probe at each of the 12 candidate locations. The design would be a modification of the illustrated concept. Our plan calls for no redundancy, but we should not completely rule out the ability to carry spares. The bus supplies the power needed by each probe during cruise and inserts itself into the proper Martian orbit, allowing them to be released at the selected locations.

3.6.4 Arrival and Deceleration. For the probes to be of any use, they must survive the impact on the Martian surface. Several deceleration techniques have

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[21] An aeroshell is the portion of a reentry vehicle designed to shed thermal build-up while inducing drag required for deorbit.

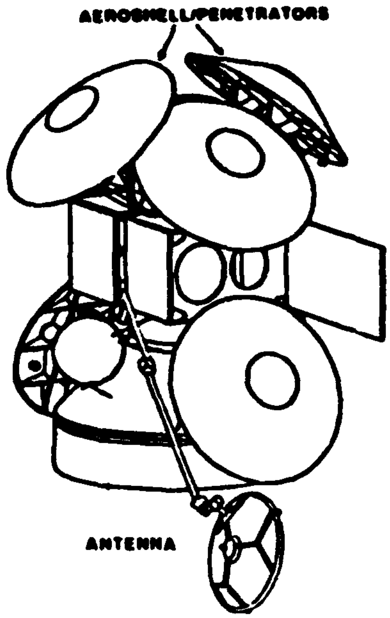

Figure 9. Aeroshell-Spacecraft Assembly Concept.

been devised the technique used driven by the type of landing desired. The three types of probe design are:

- *Penetrators.* Penetrators require the highest impact velocities as they are required to penetrate the surface to deliver their sensors.
- *Rough Landers.* Rough landers impact at speeds of 5 to 40 m/s, 10 to 50 g's.
- *Soft Landers.* Soft landers impact at speeds less than 5 m/s, less than 10 g's (21:3-4).

The landing type chosen depends upon the sensitivity of the payload. The descent to the surface can be divided into three separate deceleration regimes: high atmosphere, low atmosphere, and terminal.

3.6.4.1 High Atmosphere Deceleration. Speeds encountered in this regime range from the hypersonic to supersonic. Landers must be designed to withstand the associated aerodynamic and thermal loads. Design options include:

- *Fixed aeroshells.* These landers are bulky, and require much volume, setting a lower bound on the number of probes for a fixed-size bus.
- *Deployable aeroshells.* By easing constraints on packaging, more probes could be placed on a single bus. However these are beyond the current state of the art and further research is required into this technology.
- *Integral systems.* Integral systems such as an attached, ablative-material shell. Any material remaining from the descent could cushion impact. Again, such a design requires further research.

3.6.4.2 Low Atmosphere Deceleration. The probes will need futher retardation through the lower atmosphere. Lander survival, a function of landing site altitude is also dependent upon the deceleration method chosen. Three methods of deceleration are parachutes, ballutes, and physical shapes.

- *Parachutes.* Parachutes are well-known and proven on past missions, however, they have three inherent problems: 1) parachutes reach a terminal velocity, beyond which, further deceleration is unachievable; 2) terminal velocity does not decrease linearly with larger chutes, and large chutes encounter mass constraints necessitating a trade-off between chute mass and terminal velocity; and 3) the chute must be jettisoned at some point prior to impact to avoid covering the probe. After jettison, the probe is again subject to acceleration.
- *Ballutes.* Ballutes are large, inflatable drag inducers. They require a supply of compressed gas and are constrained by the mass and volume of the tanks. Depending upon the gas chosen, this method carries a potential for disaster.

Cost, combustibility, and environmental impact are a few of the factors that must be considered when choosing which gas to use.

- *Physical Shapes.* Landers can be designed with a low ballistic coefficient (to maximize atmospheric drag), but encounter volume constraints when doing this.

3.6.4.3 Terminal Deceleration. Terminal deceleration, or impact attenuation, is necessary to the survivability of the probe electronics and instrumentation. The more robust the payload, the higher the allowable impact speed. Mechanical parts require softer landings than solid state components. Terminal deceleration methods used in the past include:

- *Retro Rockets.* Retro rockets were proven on the Viking probes and reduced impact velocity to approximately two meters per second. The choice of retro propulsion rockets calls for an altimeter or proximity sensor to determine the point of firing. The rocket exhaust can change surface conditions, not only literally sweeping clean a small patch beneath the probe, but possibly adding combustion products to the surface constituents. Surface chemistry investigations need to take this possibility into consideration. Retro propulsion units are also subject to size and mass limitations.
- *Inflatable or Crushable Materials.* Impact-limiting cushions are proportional in size and mass to the size and mass of the probe. A small landing craft can make better use of inflatable or crushable materials.

3.6.5 Lander Design Suggestions. Burke and Mostert suggest two concepts: a two-stage penetrator, and a soft-landing, *egg* design (21:4-6).

3.6.5.1 Penetrator Concept. The penetrator, shown in Figure 10, employs a two-body separable design. The foresection penetrates beneath the surface and carries subsurface instrumentation. The tail assembly remains on the surface and performs observations in the near surface environment. The surface instrument package includes seismic triaxial accelerometers, a meteorology package, and an alpha/X-ray/proton spectrometer. Two cameras are included for descent and post-landing imaging.

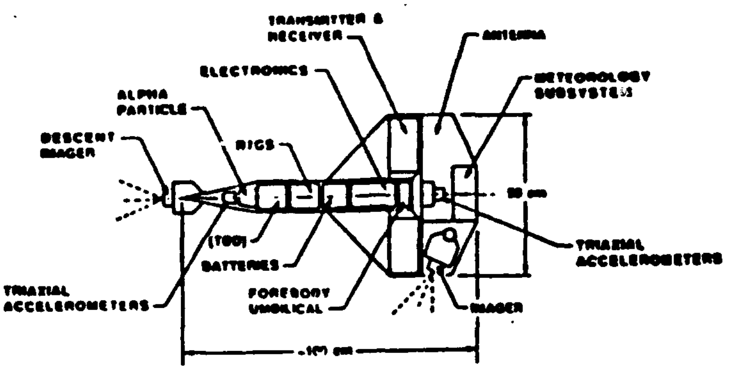
TRANSMITTER & RECEIVER
ANTENNA
ELECTRONICS
METEOROLOGY SUBSYSTEM
ALPHA PARTICLE
DESCENT IMAGER
RIGS
(TOO)
BATTERIES
TRIAXIAL ACCELEROMETERS
TRIAXIAL ACCELEROMETERS
FOREBODY UMBILICAL
IMAGER
~100 cm

Figure 10. Schematic of the Penetrator Concept.

Power is supplied by a mini-RTG and lithium batteries. For this concept to work, an RTG must be capable of surviving the high velocity impact necessary for penetration. Another design problem stems from the placement of the RTG in the foresection. Thermal dissipation from the RTG will be difficult underground. Keeping the RTG at the surface establishes a requirement for a highly survivable conducting umbilical between the sections. We also need to consider that the section remaining on the surface experiences a greater instantaneous deceleration than the penetrating section. This last consideration eliminates an alternate power source, fragile solar arrays.

High atmosphere decleration is accomplished with a fixed aeroshell. The proposed design would place two penetrators in each aeroshell for redundancy. After penetration, the penetrators are separated from the shell. A parachute can provide low atmosphere deceleration. At a set altitude, the parachute is jettisoned, allowing the penetrator to impact. Figure 11 shows the penetrator-aeroshell configuration.

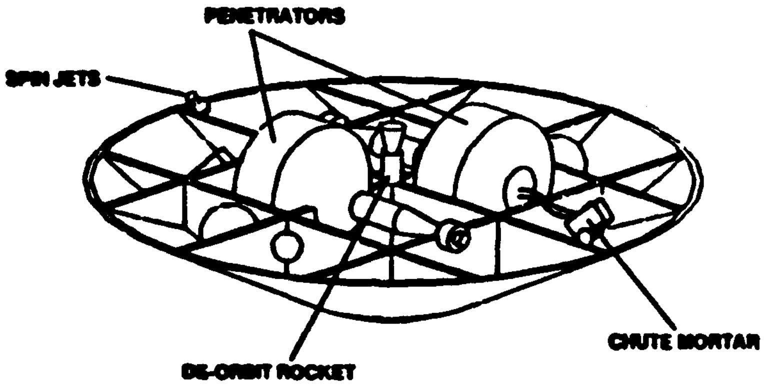

Figure 11. Penetrators as Assembled in Aeroshell.

Beyond the power difficulties, the penetrator concept has other faults that make it inappropriate for our mission. The meteorology and seismology missions would be poorly supported. As proposed, the penetrator is more suited to a mission on the order of 90 days, not two Martian years as required for Phase II. The meteorology package would also suffer from the high-impact landing. While it is simple by today's technology to produce solid-state temperature and wind sensors, we are far from producing a solid-state barometer. Pressure measurement is by nature mechanical. Making a barometer survivable would be difficult and would add bulk to the package. Lastly, the location of the package within centimeters of the surface would make the measurements subject to noise from changes in the near-surface micro-environment. More representative measurements could be made with a boom extending the meteorological package to about 1.5 meters above the surface, but this will add mechanical mass to the system.

3.6.5.2 Soft-Lander (Egg) Concept. The soft-lander (egg) design is more suited to our mission requirements. Inspired by the Soviet Luna 9 module, this design is shown in its closed, transportable configuration in Figure 12. The design has four petals, each carrying different instrumentation, which open upon successful landing. Figure 14 shows the egg in its full-open operational configuration.

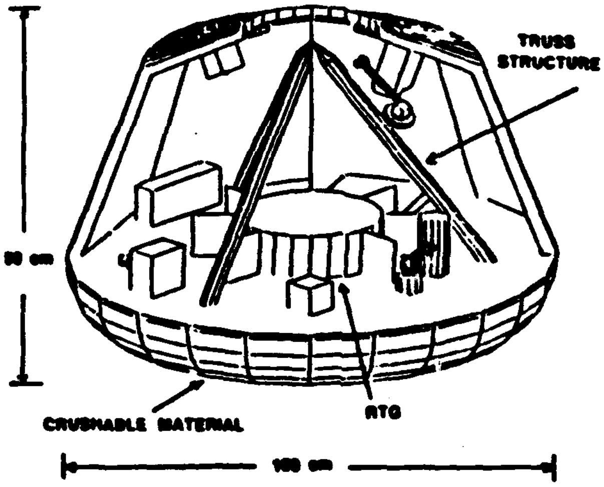
TRUSS STRUCTURE
CRUSHABLE MATERIAL
RTG

Figure 12. Egg Probe in its Closed Configuration.

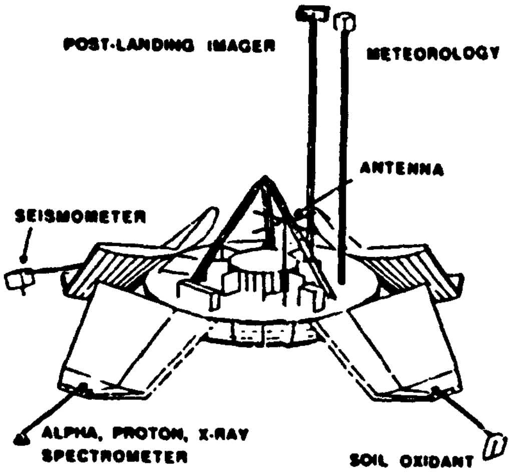
POST-LANDING IMAGER
METEOROLOGY
ANTENNA
SEISMOMETER
ALPHA, PROTON, X-RAY SPECTROMETER
SOIL OXIDANT

Figure 13. Egg Probe in its Open, Operational Configuration.

Deceleration. In its closed configuration the egg is attached to a fixed aeroshell and attached to the dispenser-bus. Figure 13 shows the egg within its aeroshell. The bus must be capable of carrying several such packages, and inserting them from orbit. The aeroshell supplies initial deceleration and shields the egg during entry. A parachute provides low atmosphere deceleration. After jettisoning its parachute at a set altitude, the probe uses proximity sensors to fire its retro-rocket system. With retro-rocket assist the probe makes its soft landing at only a few meters per second. For one last measure of softness, the entire outer surface of the egg carries crushable materials.

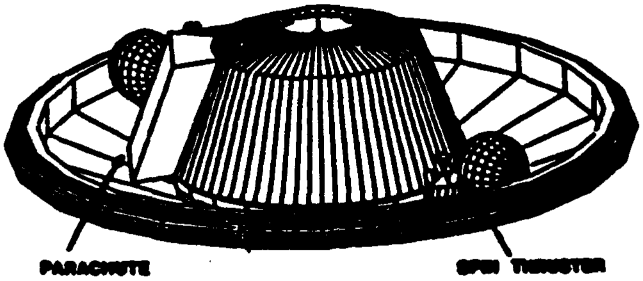

Figure 14. Egg Probe Within its Aeroshell.

Instrumentation. The instruments and their support and deployment structure do not need a robust design. Design resources can instead be channeled into redundancy and longevity. The boom for the meteorology package ensures that measurements are representative of the environment encountered by a human standing erect. The unfolded petal design will also deploy the chemical samplers outside of the region disturbed by the landing. The post-landing imager, atop its boom, provides a wider view of the landing site than that on the penetrator.

The egg concept carries a seismometer, a differential scanning calorimeter and evolved gas analyzer (DSC-EGA) for elemental identification, a soil oxidant experiment, and a meteorology package, among other instruments. Design specifics for the instrumentation, as provided by Mostert and Knocke, are given in Table 2 (64:4).

Table 2. Design Specifics for the Egg Lander Instrumentation (Strawman Payload).

Instrument	Mass grams	Power mW	Data Rate	Operations
Meteorology Package	300	75	40 bits/hr	continuous
Seismometer	500	300	1.25 Mbit/event	4 events/day
Alpha,Proton,X-ray, Spectrometer	250	150	12 kbit/meas.	TBD
Soil Oxidants	1000	1000	12 kbit/meas.	TBD
Accelerometer	150	1000	15 kbps	2 seconds
Descent Imager	400	1000	2 Mbits/event	10 events
Post-Landing Imager	400	1000	2 Mbits/event	TBD (events)

Table 3. Subsurface Science.

Instrument	Mass grams	Pk Power mW	Data Rate	Operations
Gamma-ray Spectrometer	1300	800	256 kbit/sample	once/6 hrs
DSC	1200	15000	64 bps	TBD
EGA	3600	7000	64 bps	TBD
Neutron Spectrometer	1300	800	1 kbit/meas.	once/6 hrs
Temperature	50	50	1 bps	TBD

The meteorology package will use technology proven effective in the Martian environment on the Viking missions (41:4559), with only minor modifications to bring the system up to current standards and increase longevity. The ambient temperature sensor consists of three Chromel-Constantan thermocouples wired in parallel. Measurements over the entire range of expected Martian temperatures are achieved within about 1.5 degrees Celsius. Two hot film (platinum) sensors are mounted at right angles in the horizontal plane to measure windspeed (based on differential cooling). The films are maintained at 100 degrees Celsius above ambient temperature by a reference temperature sensor. Windspeed accuracies within ten percent have been achieved in laboratory conditions over a range of 2 to 150 m/s. The wind sensor also gives wind direction but the quadrant is ambiguous. A quadrant sensor selects the proper quadrant. This sensor consists of a heated cylindrical core surrounded by four thermocouple junctions at equal angles and distance from the core. The thermocouples measure the wake from the heated core. Wind direction can be determined to within 10 degrees of azimuth. Ambient pressure is measured with a Kiel barometer (stressed diaphragm type) with accuracy better than 0.09 mbar over the range of 0 to 20 mbar. Figures 15 – 18 illustrate these instruments as deployed upon the Viking spacecrafts.

Communications. **Each probe will carry a communications package and will need software to compress and transmit data to the communications satellite. A concept for future consideration is a failsafe beacon which will tell the orbiting bus whether the egg landed safely. If the orbiting bus is capable of carrying spares, it could then dispense a replacement for a broken egg.**

Mass Estimate. **Mostert and Knocke estimate the total mass of each egg and aeroshell as 62.00 and 59.88 kg, respectively (64:6). A breakdown by subsystem is offered in Table 4.**

Table 4. Mass Estimate for Egg Probes.

Subsystem	Mass kg	Subsystem	Mass kg
Communications		Aeroshell	
Transmitter/Receiver	2.00	Structure	20.00
Antenna	0.40	Ablator	18.20
		Separation system	2.00
C & DH	3.00	Parachute mortar	1.00
		De-orbit instrumentation	1.00
Thermal Control			
Heater	0.10	De-orbit engine	2.50
Insulation	2.00	De-orbit tanks and structure	0.70
		Separation and spin system	1.50
Structure			
Primary structure	17.50	De-orbit fuel	1.00
Mechanisms	2.00		
Parachute	5.00		
Retro rocket	5.00		
Impact absorber	4.00		
Power			
RTG	5.00		
Batteries	1.00		
Science Payload	2.60		
Contingency (25 percent)	12.40	Contingency (25 percent)	11.98
Subtotal	62.00	Subtotal	59.88
Total	121.88		

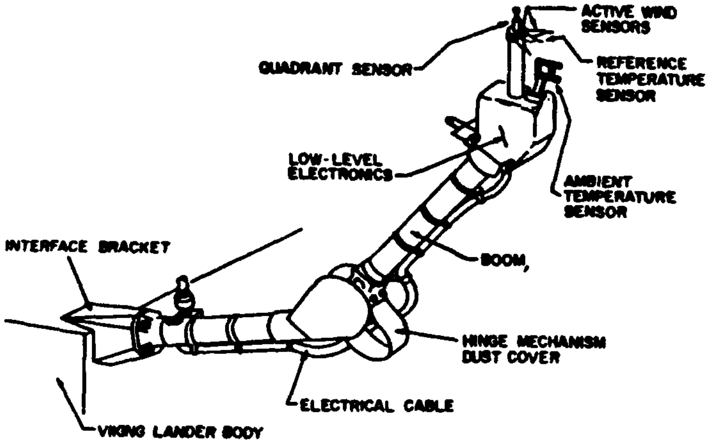
ACTIVE WIND SENSORS
QUADRANT SENSOR
REFERENCE TEMPERATURE SENSOR
LOW-LEVEL ELECTRONICS
AMBIENT TEMPERATURE SENSOR
INTERFACE BRACKET
BOOM,
HINGE MECHANISM DUST COVER
ELECTRICAL CABLE
VIKING LANDER BODY

Figure 15. Viking Meteorology Boom (22:1099).

3.6.6 Summary. The egg concept is fully capable of completing both our scientific and site selection missions. Based on its reliance on proven technology and with only minor improvements, the egg concept could have a long productive lifetime. This concept is easily adapted to other instrument packages and could be used for exploration of other bodies as well as follow-on missions to Mars. The egg not only fulfills our mission, but also meets our goal of more modular, interchangeable technology.

3.7 Martian Rover-Lander.

This section describes the objectives and technical requirements for a roving laboratory on the Martian surface. There is little precedent for this device. A component-level design was performed in order to estimate the requirements levied on the other Phase II components (primarily MTV and MaRCoS), and to validate the Rover-lander concept. As there is sparse literature on this subject, much of that speculative, the proposed design and operational concept draws significantly on the Viking missions and known environmental conditions.

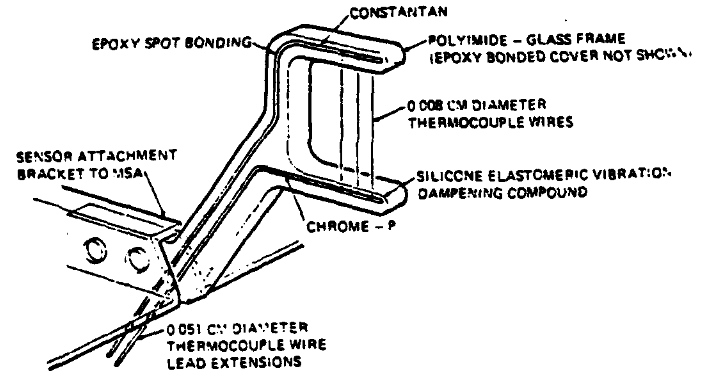
CONSTANTAN
EPOXY SPOT BONDING
POLYIMIDE – GLASS FRAME (EPOXY BONDED COVER NOT SHOWN)
0.008 CM DIAMETER THERMOCOUPLE WIRES
SENSOR ATTACHMENT BRACKET TO MSA
SILICONE ELASTOMERIC VIBRATION DAMPENING COMPOUND
CHROME – P
0.051 CM DIAMETER THERMOCOUPLE WIRE LEAD EXTENSIONS

Figure 16. Viking Temperature Sensor (22:1098).

3.7.1 Mission Objectives. The primary mission of this phase of the Project Ares is the gathering of data needed to make the final decisions regarding future manned missions. The data relate to the hazards presented by the Martian environment. The following critical questions need to be addressed:

- Can the explorers and their equipment survive a landing on the Martian surface?
- Can explorers survive on the Martian surface for short durations (30 days)?
- Do the planned missions present a long-term hazard to the health of the explorers?
- How can the environmental hazards posed by landing and working on the Martian surface be minimized?

Every effort must be made to fully understand the interplanetary and Martian environments before we begin a manned mission. The known hazards of equipment failure or chance collision with debris are significant enough without subjecting the explorers to the dangers of the unknown. Unmanned probes have allowed us to characterize the near-Martian space, and to a small degree, the Martian surface. More

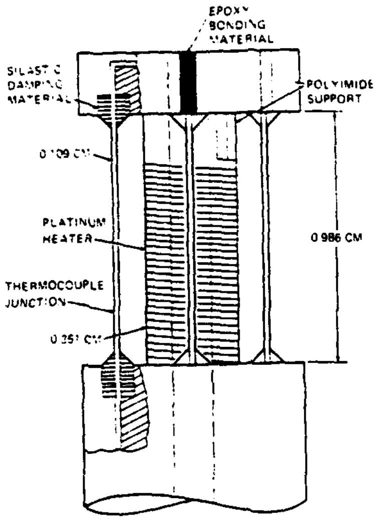
EPOXY BONDING MATERIAL
SILASTIC DAMPING MATERIAL
POLYIMIDE SUPPORT
0.109 CM
PLATINUM HEATER
0.986 CM
THERMOCOUPLE JUNCTION
0.251 CM

Figure 17. Viking Quadrant Sensor (22:1098).

detailed information is required to evaluate the exact conditions humans will find at their specific landing site (provided a suitable site can even be found). Detailed requirements assigned to this manned-mission precursor are described below.

3.7.1.1 Environmental Hazard Evaluation. Aside from the hazards of landing and surviving in the cold and hostile Martian environment, chemical and biological poisons could pose serious hazards to explorers. Fortunately, Mars has sufficient atmosphere and planetary magnetic field to protect the surface from all but the strongest solar radiation (60:137). Current theory, supported by somewhat ambiguous data from the Viking Lander microbiological experiments, indicates no life is present, but further tests are needed (44:4659).

3.7.1.2 Manned Landing Site Certification. This mission must explore and return first-person data on the best landing site chosen using the MSM data. From this information the final pinpoint location for the landing will be determined. The area must be explored in detail, allowing the manned mission to concentrate on other tasks after arrival.

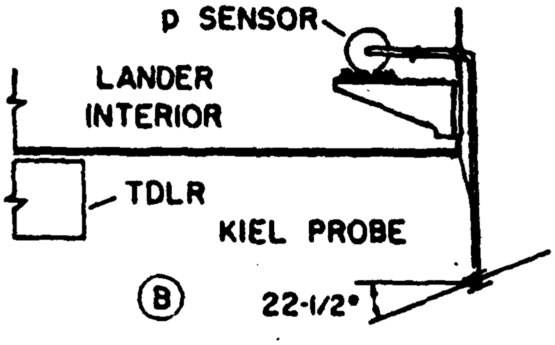

Figure 18. Kiel Probe Barometer as Deployed on Viking (76:4363).

3.7.1.3 Long-Term Resource Evaluation. This mission must also perform experiments to evaluate the potential for long-term manned presence on Mars. These experiments will help determine if water is extractable from the environment, whether surface materials can be used for building material or chemical processing, and whether long-term hazards exist to preclude a continuing manned presence.

3.7.2 Mission Requirements. These are the specific requirements pertaining to the manned follow-on missions that must be addressed by experiments on the mobile laboratory.

3.7.2.1 Surface Physical Characteristics. The load bearing characteristics and other mechanical properties of the Martian surface will be needed to accurately predict the stresses on landing craft. In addition, data will also feed into the design of structures and shelters requiring surface anchoring.

Dust Properties. Knowledge of the shape and size of the ubiquitous Martian dust particles will be important to describe its abrasive effects. Fixed structures of all types will be exposed to the erosive effect of aerosol particles, and the dust will permeate mechanical joints and seams.

Suitability for Landing. After the soil density and load-bearing capability is measured, the last physical impediment is finding a suitable landing site. The general characteristics of the optimal site include:

- Flat terrain for 5-20 km around the site.
- Relative freedom from rocks and boulders larger than .25 m.
- Proximity to variegated surface features of interest.
- Proximity to exploitable resources.

3.7.2.2 Suitability for Manned Exploration/Colonization.

In later missions, there will be trade-offs among landing site hazards, site exploratory and site scientific value. Variety of nearby physical features (terrain, mineral deposits, et cetera.) will become more important. Local variations in background radiation, if present, will also affect permanent base site selection, due to the danger of increased long-term exposure. The mobile laboratory must be equipped to determine these characteristics.

3.7.2.3 Surface Chemical Characteristics.

Data on the toxicity, water content, and mineral composition of the dust and soil at and around the landing site will also be needed to make decisions regarding manned operations.

Toxicity. Special equipment and operational techniques will be required if the Martian soil contains substances hazardous to explorers. Decontamination and/or isolation procedures will be needed, not only on return to Earth, but more importantly during daily operations on the Martian surface. The two primary hazards are from heavy metals (such as cadmium and mercury) and bioactive substances (38).

Water Content. Of all the exploitable resources known to be available on Mars, water in any form is the most important. Detailed analysis of all possible water sources must be performed to make long-term decisions regarding the permanent human presence on Mars. These sources include water absorbed by the dust and soil, water of hydration chemically fixed in the surface materials, sub-surface ice and permafrost, and surface ice near the northern Martian pole (48).

Detailed Mineralogical Assay. In addition to water, other minerals could be used by permanent habitants to synthesize needed compounds or construct building materials. Experiments to determine not just elemental constituents but also chemical compounds are needed to characterize these minerals.

3.7.3 Concept of Operations. There will be two rover-lander packages, one for each of the two sites deemed optimum from the data returned by the previous missions. The rover is a mobile laboratory. The rover's mission is to ensure Mars is suitable for man. The lander's duty is to safely convey the rover from orbit to the Martian surface. In addition, the lander will perform some scientific investigations on its own. Both rovers will be active on Mars simultaneously. Communications will be direct to the MaRCoS in stationary orbits.

3.7.3.1 Earth Launch. Both rover-lander packages will be launched at the same time on a common bus vehicle.

3.7.3.2 Mars Arrival. The transport system will place the landers in Mars orbit. The orbital altitude will be 200 km, and the minimum orbit inclination will be equal to the maximum landing site latitude. After arrival, the vehicles will remain in orbit up to six months for:

- System checkout/operational validation.
- Communications link establishment with MaRCoS.
- Final site survey, checking for weather and surface conditions (using the MSM and probe data).
- Descent preparation (deployment from bus,configuration changes, et cetera).

3.7.3.3 Deorbit. Obeying commands preloaded during descent preparation, the lander will perform a deorbit burn.

3.7.3.4 Descent. The attitude of the lander at deorbit will cause the ablative base of the aeroshell to be presented to the thickening atmosphere first, protecting the lander from frictional heating. After a fixed interval, the descent parachute will deploy and the aeroshell base will drop away, exposing the lander braking motor, radar altimeter, and science sensors. The chute will slow the descent of the lander to 50 meters per second. The lander science package will be activated, allowing the measurement of meteorological data during the slowed descent. The rover video system will also be activated, enabling its camera to view the surface during the descent while in the stored position on the rover.

3.7.3.5 Prelanding. The doppler radar altimeter will cue the start time and duration of the braking motor firing, which will slow the lander descent to below one meter per second. Radar data and thrust vectoring will be used to reduce the lander's lateral velocity and rotation rate to near zero, and to maintain attitude. The descent parachute and remainder of the aeroshell will separate upon confirmation of the start of braking motor firing. The landing legs will then be deployed.

3.7.3.6 Landing. Collapsible, shock-absorbing pads on the four lander *feet* reduce the stress of the touchdown. The legs will be adjustable (up to .5 m) to level the lander and compensate for sloping terrain or uneven surface features. After touchdown, the rover communications package will deploy and acquire the MaRCoS system to allow high-rate data transmission. A complete diagnostic analysis of the rover-lander subsystems will again be performed. A survey of the immediate vicinity of the lander will be conducted (using the rover video system) to begin the rover deployment decision-making process. The applicable rover science packages will be activated and begin data gathering. The rover will remain on-board the lander until:

1) the performance of all rover systems have been evaluated, 2) the local surface conditions (wind velocity, dust count, temperature) have been measured and are appropriate, and 3) an operational strategy has been determined and the necessary operational commands stored on the rover and lander.

3.7.3.7 Rover Deployment. Just prior to deployment, the lander attitude will be adjusted as necessary. A ramp will deploy from beneath the rover platform on the lander on the side determined to be optimum. The operation will be observed by the rover video system, and sensors in the ramp will confirm the ramp is fully extended, on firm ground, and is adequately level. Once the ramp safety is confirmed by multiple sources, the rover will switch to fully autonomous operation, disconnect from the lander and its Rover Support System, store any sensors or antennas that would impede egress from the lander, and roll down the ramp.

3.7.4 Rover-Lander Operations. The rover will move in discrete, individually commanded segments. The Earth-bound mission controllers will plan movement operations based on previously returned rover video and data, then upload movement programs. The Rover Support Subsystem Controller will have a brilliant watchdog routine that continuously monitors speed, attitude, and orientation. The watchdog will override stored commands and preempt rover operations if critical vehicle parameters reach set limits. Once it has control, the watchdog will act according to stored program until the vehicle is out of danger, then abort the movements orders and return control to mission operators. Rover navigation will be performed using accurate and detailed maps provided by MSM compared against the video returned by the rover. A transmitter beacon (identical to the beacon on the probes) will assist in pinpointing the rover position.

3.7.4.1 Mission Performance. Operation of the Science Subsystem Controller for experimental package operations will be similar to the Support Controller, with the controller operating in a hybrid commanded/autonomous mode. The full suite of rover experiments will be performed within meters of the deployment site and within hours of deployment to the surface, with the results stored and transmitted near-real-time to Earth. The lander will continue to provide low-rate scientific data (through the MaRCoS) after the rover has departed.

3.7.5 Technical Description of Rover. Two rovers will be conveyed to the Martian surface on landers, which support and protect the rovers during atmosphere entry and landing. The rover landing sites will be the two sites deemed most favorable for a manned landing by examination of the mapper and probe data. Because of its mechanical complexity and limited store of experiment consumables, the rover will have a planned operational lifetime of one (Earth) year. The rover's mass is approximately 430 kg. Its footprint is two meters by two meters including wheels; the chassis dimensions are one meter wide by two meters long. The chassis supports the science and support subsystems. A summary of critical design parameters in shown in Table 5.

3.7.5.1 Science Subsystem. The science subsystem has the responsibility for answering the questions posed by the requirements. There are two major subsystems: the Experiment Package (EP) and Material Handling Assembly (MHA).

3.7.5.2 Experiment Package. This subsystem performs the physical and chemical analysis of the Martian surface. The major components are:

- *Advanced Mass Spectrometer.* This instrument will determine the elemental composition of gases either vented in from the Martian atmosphere or transferred in from Soil Decomposition Experiment.
- *Advanced Multispectral Flourescence Spectrometer.* This device exposes the materials being examined to ionizing radiation of various energies (alpha par-

Table 5. Rover Total Parameters.

	Mass kg	Volume m^3	Inactive Power W	Active Power W	Data/ Day kbits
Rover (Total)	430	4.07838	6.1	62.3	50207
Science Subsystem	54	0.011755	3.3	35.2	35562
Experiment Package	20	0.006255	2.9	20.2	34440
Adv Mass Spectrometer	2	0.001	0.1	2	30
Adv Fluorescence Spectrometer	2	0.001	0.1	2	30
Imaging System	1	0.00025	0.1	2	12000
Microscopic Examination Module	2	0.001	0.1	0.2	1000
Soil Decomposition Experiment	5	0.001	0.2	10	100
Accelerometer	1	0.000005	0.1	1	1440
Seismometer	1	0.0005	0.1	0.5	4000
Meteorology	1	0.0005	0.1	0.5	1440
Science Process Controller	5	0.001	2	2	14400
Material Handling Assembly	34	0.0055	0.4	15	1122
Sample Distribution System	3	0.002	0.1	1	30
Corer	20	0.002	0.1	10	20
Atmospheric Particulate Sampler	1	0.0005	0.1	1	72
Manipulator Arm	10	0.001	0.1	3	1000
Support Subsystems	376	4.066625	2.8	27.1	14645
Structure	100	.001	N/A	N/A	N/A
Undercarriage	200	4	0.5	20	100
Power Generation and Distribution	150	0.0625	0	0	72
Communications	20	0.003	0.2	5	72
Rover Navigational Beacon	1	0.000125	0.1	0.1	1
Support Process Controller	5	0.001	2	2	14400

ticles, protons, X-Rays, gamma rays) and examines the resulting reradiation for indications of composition.

- *Video System.* This consists of a single, multipurpose imaging system. The video camera is mounted on a boom capable of raising the camera two meters above the top of the rover. From that (or any intermediate) position, the camera can swivel 360 degrees. In the nominal stowed position, the camera is connected via fiber optics to the Microscopic Examination Module. In addition, through the Rover Support Interface, the camera is able to image the rover-lander descent (reference Section 3.7.7.6).
- *Microscopic Examination Module (MEM).* The MEM operates in conjunction with the Video Subsystem to allow detailed optical examination of Martian materials. An examination platform illuminated by filtered and polarized light is scanned by a lens system connected to the video camera (in its stowed position) via dense optical fibers. Highly informative crystalline structures, dust particle size and shape distribution, and perhaps remnants of long-dead lifeforms are the rationales for this experiment.
- *Soil Decomposition Experiment.* This experiment exposes the Martian materials to a variety of liquid chemical agents. The resulting gases (and liquids flashed into gas) are transferred to the mass spectrometer for analysis. The characteristics of the experiment by-products allows the compounds in the original materials to be deduced.
- *Meteorological/Physical Experiments.* This catch-all package includes the standard meteorological and seismographical instrument set included on the probes, and adds an accelerometer, a magnetometer, and a radiation dosimeter. The magnetometer and dosimeter are mounted on an extendable 3-meter boom to reduce the effects of the other rover apparatus on their readings.
- *Science Process Controller.* This is the electronic computer that commands and controls the EP and the MHA (described in the next section). A minimum

of 100 megabytes of data storage is required to buffer experiment results until transmitted to MaRCoS.

3.7.5.3 Material Handling Assembly. This subsystem will retrieve materials from the Martian subsoil, surface, and atmosphere. It then measures the volume and mass of the sample (if needed) and conveys it to the selected experiment. This system also takes the gaseous output from the different chambers of the decomposition experiment and transports them to the mass spectrometer. Facilities are included for self-cleaning and purging to avoid inter-contamination of the various samples and experiment by-products. Its components are an articulated manipulator (henceforth, arm), a core sampler, a dust collector, and means of moving samples around inside the rover. This assembly is commanded by the Science Process Controller.

- *Sample Distribution System (SDS).* The SDS takes samples presented to the system by the arm and conveys them to the requested experiment. There is a hatch in the rover cover with a funnel underneath. When a sample has been collected for analysis by the arm, a small car that moves between the various experiment stations is commanded under the funnel. The hatch opens and the arm drops the sample. The car mechanism measures the sample mass and volume, then takes the sample to the requested experiment.
- *Corer.* A three meter long, two centimeter diameter core sampler is mounted on the rear of the rover external case. It is constructed of six hollow, concentric, interlocking 0.5 meter tubes, with the innermost tube equipped with a boring bit. The unit telescopes to less than one meter in length in the stowed position. In operation, the corer will rotate and extend the innermost segment downward its full 0.5 meter length. This length will then be retracted, the rover will move backward a predefined distance, and, with the rover still moving slowly, the borer tube opened and the sample emptied on the cold Martian soil. The

rover then moves forward and sites (or feels for) and aligns with the hole it just made. The borer extends and bores until another 0.5 meter of material is collected. The process is repeated, each sample being laid out slightly further back than its predecessor. When the full three meters is collected, the Rover stows the corer and drives (is commanded) to a spot where the video imager can examine the results of the dig. The vise hand on the arm will be directed to gather a small subsurface sample and convey it to the SDS.

- *Atmospheric Particulate Sampler (APS).* The APS is a small vacuum similar to a dust-buster with a fine-mesh metal screen. The APS will be activated during the Martian night to avoid collecting dust inadvertently kicked up by rover activity. The arm will remove the reusable filter and empty it in the SDS.
- *Manipulator Arm.* An extendable two-meter long articulated arm for retrieving surface samples. The hand consists of a simple scoop similar to a miniature crane bucket, and a vise capable of both gently lifting samples or (with increased pressure) crumbling small rocks for later scooping. The arm retrieves the contents of the APS, and the SDS samples the subsurface material made available by the corer.

3.7.6 Support Subsystems. These components transport, shelter, power, and provide communication for the science system. The rover structure provides for thermal control and power distribution for the other subassemblies.

3.7.6.1 Undercarriage. The undercarriage is the chassis on which rests the remainder of the rover systems. The carriage consists primarily of a two square meter support platform and six electrically-driven oversized tires, three on a side (54:328). Two redundant, fifteen-watt motors drive each wheel, and steering is accomplished using differential wheel velocity. The 0.5 meter diameter tires with conical inner rims will allow the rover to overcome 0.25 meter high steps and free-flowing slopes up to 35 degrees of inclination.

3.7.6.2 Power Generation and Distribution. Electrical power for the rover is provided by a radioisotope thermoelectric generator (RTG), supplying 100 watts.

3.7.6.3 Communications. Primary communications is direct to the MaRCoS via a one-half meter diameter, C-band (5 GHz) steerable dish. Communications will not be performed while rover is in motion. Pointing information is derived from the stored MaRCoS azimuth/elevation and the rover attitude. An autoseeking algorithm using the MaRCoS carrier provides final fine-tuning of the communications channel. If MaRCoS is unavailable, the system can provide direct-to-Earth two-way communications at a significantly reduced data rate.

3.7.6.4 Rover Navigational Beacon. The rover has a low-power omnidirectional beacon that, in conjunction with the beacon receiver on mapper, provides a secondary locator method via doppler analysis of the received signal on the mapper (this beacon is identical to the beacon on the probes). The beacon is powered directly by the RTG and precluding failure, will transmit well after the operational life of the rover is over, providing a landing reference for future missions in its vicinity.

3.7.6.5 Support Process Controller. This computer commands and controls the above support elements. As with all the rover-lander controllers, it operates in a semi-autonomous mode using transient and permanent stored programs. The controller automatically diagnosis adverse conditions, interrupts the transient program, and takes full control of rover operations to eliminate or at least mitigate the hazardous situation. The controller is also responsible for the collection and insertion of rover state-of-health telemetry into the scientific data stream to MaRCoS.

3.7.7 Technical Description of Lander. The less complex and stationary lander's operational life will be ten years (the limit of usable RTG power). The major subsystems of the lander are the Descent/Landing, Communications, Rover Support Interface, Lander Science, and the Lander Controller. The lander technical parameters are listed in Table 6.

3.7.7.1 Descent/Landing Subsystem. This subsystem serves to bring the entire rover-lander package from orbit to a soft landing on Mars.

- *Structure.* The lander main structure is a 2.5 m by 2.5 m platform, on which rests the rover, held rigidly in place by clamps and explosive bolts. The platform, 0.5 meter thick, contains the rover deployment mechanism and the remaining lander subsystems. Four legs are attached to the corners of the platform and are folded against the sides in the stowed position. The deployed legs will absorb the landing shock, provide clearance for the lander platform above the Martian surface, and actively level the lander platform after landing. Thermal control is provided by this subsystem for the rover-lander.
- *Aeroshell.* The shell is a protective cover completely surrounding the rover- lander assembly. The base is a spheroidal section of heat-ablative material three meters in diameter. It will protect the assembly from the heat of atmospheric entry, allow a stable and predictable trajectory through the atmosphere, and provide some lift (81:3963). The base is jettisoned just after parachute deployment. The top cover is a rigid frame covered with insulating material, arching over the parked rover. It supports the descent parachute and stores the hydrazine fuel used for the deorbit burn. The frame is rigidly attached to the lander with a decoupling mechanism and will be jettisoned just before the braking motor fires. Ports in the top cover and associated ducts vent a portion of lander braking motor thrust to allow the lander controller to adjust the vehicle orientation and trajectory during orbit, deorbit, and entry.
- *Descent Parachute.* The second velocity reduction method is a 20-meter diameter conventional parachute, attached to the top of the aeroshell frame (81:3964).

Table 6. Lander Parameters.

	Mass kg	Volume m^3	Inactive Power W	Active Power W	Data/ Day kbits
On-orbit	624	2.36375	2.7	7	1512
Descent	624	2.28575	2.9	8.4	7668
Pre-deployment	314	0.28775	2.9	18.1	7659
Post-deployment	314	0.005	2.4	8	6952
Descent/Landing Subsystem	370	2.156	0.4	0.4	84
Structure	100	0.002	0	0	0
Aeroshell	100	2	0.1	0.1	1
Descent Parachute	30	0.075	0.1	0.1	1
Braking Motor	40	0.004	0.1	0.1	10
Deorbit Fuel	100	0.075	0.1	0.1	72
Braking Fuel	200	0.15	0.1	0.1	0
Communications	20	0.003	0.2	5	72
Rover Support Subsystem	227	0.20375	0.3	11.1	6147
Structure	200	0.001	N/A	N/A	N/A
Rover Support Interface	5	0.001	0.1	1	72
Rover Deployment Mechanism	20	0.2	0.1	10	72
Rover Navigation Beacon	1	0.00025	0.1	0.1	3
Descent Imaging System	1	0.0025	0	0	6000
Lander Science Subsystem	2	0.001	0.2	1	5440
Seismograph	1	0.0005	0.1	0.5	4000
Meteorological	1	0.0005	0.1	0.5	1440
Lander Controller	5	0.001	2	2	1440

The parachute will carry away the aeroshell top seconds prior to braking motor firing.

- *Braking Motor.* The lander braking motor will slow the vehicle to its final landing velocity, less than one meter per second. It consists of two clusters of four individually-controlled, thrust-vectored hydrazine rocket motors. While the aeroshell is in place, two motors from each cluster are ducted through the upper shell to provide attitude control and deorbit thrust. The multi-use braking motor is managed by the lander controller in all phases of landing operations.

3.7.7.2 Communications. While in orbit, during descent, and after landing, the lander communications will be directly to the MaRCoS. The lander has two low-gain, C-band omnidirectional antennas to provide hemispherical coverage above the lander platform. This is acceptable due to the lander's relatively low data transmission/reception requirements and the lack of requirement for direct-to-Earth contingency communications. (The rover communications subsystem provides that capability until it is deployed.)

3.7.7.3 Rover Support Interface. This is the interconnection between the rover and lander that allows them to share resources while they are mated. The lander controller is in charge until deployment, operating the rover systems through the rover controllers. The primary use of this subsystem is to allow the lander to use the rover high-gain parabolic antenna during Mars-orbit contingency operations (direct-to-Earth, MaRCoS unavailable) and during rover deployment preparation (when the rover systems are active).

3.7.7.4 Lander Science Subsystem. This package contains the seismographic meteorological instrumentation common to the rover and probes. The measurements are uplinked to MaRCoS via the low-rate C-band antennas.

Table 7. Rover-Lander Total Parameters.

Rover-Lander Total	Mass (kg)	Volume (m^3)	Inactive Power (W)	Active Power (W)	Data/ Day (kbits)
On-orbit	1054	6.44213	8.8	69.3	30312
Descent	1054	6.36413	9	70.7	1512
Deployment	744	4.36613	9	80.4	57866

3.7.7.5 Lander Controller. This controller manages the on-orbit, landing, and rover predeployment operational phases. Before rover deployment, it commands the rover process controllers and uses the rover communication system to relay scientific and state-of-health data to MaRCoS.

3.7.7.6 Rover Deployment Mechanism. The upper section of the lander platform (just beneath the rover) deploys a two meter ramp from either end of the lander. The ramp is sectioned to adjust for minor angular differences uncompensated for by the landing legs. The ramp is extended, lowered to the surface, flexed to conform to the angle difference, then finally made rigid to support the weight of the departing rover.

3.7.7.7 Lander Navigation Beacon. This is a continuous, low-power beacon in the UHF band, identical to that on the rover and probes except for the exact frequency (reference Section 3.7.6.4).

3.7.7.8 Descent Imaging System. The decent imaging system uses the Rover Imager in its stowed position to capture of images during descent, to provide landing location validation and detailed aerial view of the surface near the lander.

3.7.8 Rover-Lander Interface Parameters. Table 7 shows the critical parameters needed to estimate the loads placed on the MTV and its data interface rate with the MaRCoS.

3.7.9 Critical Factors and Assumptions. Here are the assumptions and external factors necessary in developing the operational techniques and technological capabilities for this stage of the mission.

3.7.9.1 Advanced Experiments. Research uncovered no documentation on the specifications for the advanced scientific payload on the rover. The principle assumption is that a total package powerful enough to perform a complete chemical assay can be small and light enough to mount on a mobile platform. The requirements, however, insist on an analysis-in-depth, and the only viable alternative is a *soil return* mission. Bringing materials back to Earth would allow the best analysis possible. This concept was dropped in favor of the rover-lander because:

- Including a large, heavy, Earth-return motor segment on the Lander would severely complicate the Mars landing operation.
- If the return motor remains in orbit, a smaller lift-to-orbit motor must still be included on the lander, and the return package must reach orbit and rendezvous, a complex and hazardous process when performed remotely.
- The long exposure of the Martian materials to the interplanetary environment will change their characteristics, nullifying some of the advantage of returning the material to Earth.
- The additional weight on the return system would probably preclude the use of a mobile robot, effectively trading off the capability to perform manned landing site reconnaissance in favor of a more detailed soil analysis. This safety versus science trade-off is inappropriate for this early exploration phase.

3.7.9.2 Parameters. The rover-lander mission requirements drove the overall system design described above. Some subassemblies were envisioned solely for this project, while others were taken from outside sources. In many cases, research findings suggested the direction of the design, without supplying data that was directly attributable in this report. These sources include after-action wish-lists suggested in the reports of Viking researchers Banin (11) and Gooding (38), and proposals for similar devices like static landers (Burke (21) and Mostert and Knocke (64)). Research uncovered only speculation regarding alternative rover designs and could not supply us any design parameters. These parameters are needed to validate the practicality of the proposed rover and provide interface data to the MaRCoS and MTV design teams. Three methods used to derive the rover-lander parameter were speculative proposals, extrapolation from Viking Design Data, and logical deduction:

Speculative Proposals. JPL researchers proposed experiments for a lander network that were of the same type planned for this project (21) (64). Their estimated parameters for the scientific experiment boxes were useful for order-of-magnitude comparisons to the parameters for Rover science packages. Kemurdzhian's very general description of the Soviet Lunokhod and Marsokhod planetary rover platforms filled in some undercarriage details (54).

Extrapolation from Viking Design Data. Viking data, gleaned from Soffen (81), Clark (24), and Horowitz (44), was useful in determining the type of scientific hardware on the rovers and landers, what the operational techniques for these types of experiments should be, and general Viking operational data.

Logical Deduction. In cases where there was no information, estimates were made based on similar current technology. This was the technique used to arrive at the subcomponent masses and volumes. In every case, the estimates are very conservative, in keeping with the primary purpose of this report, concept validation.

3.7.10 Summary. Based on the above analysis, including research data and reasonable estimates of rover-lander performance and parameters, this portion of Project Ares is feasible. Provided sufficient resources are dedicated to rover-lander design and fabrication, there are no significant technological obstacles to a proposed 2005 launch date.

3.8 Mission Platform Design

Now that each of the major mission systems have been analyzed, we will now discuss the platform subsystems which support them. The payloads are of two types, high orbit and low orbit systems. In the following paragraphs we describe the design choices and decisions involved in building the spacecraft bus.

3.8.1 Interplanetary Transport. The long journey to Mars will be a relatively calm period of operations for Phase II mission systems. This section describes the equipment status and operational modes during the 200 to 300 day voyage in interplanetary space. Mission systems which include MaRCoS I and II, the MSM, a platform for inserting surface probes (probe dispenser), and the rover-lander assemblies are referred to as payloads in relation to the MTV. The MTV is expected to provide minimal power and communications to each payload during the journey to maintain system operations and telemetry and commanding capabilities. Near the termination of the transport, each payload will separate from the MTV and proceed to its operational location on its own accord. The details of deep-space storage and orbit insertion are explained further below.

While in storage on the MTV, each payload will have its antennas, solar arrays, and other deployable mechanisms stowed. The electrical systems are to operate at survival power levels which aids in preventing premature failure. Since exact power levels are difficult to predict, the power required from the MTV is set at twice that expected. Survival power levels are estimated as 250 watts per satellite[22] which makes a total of 500 watts of electrical power required by each satellite while attached to the MTV. Minimum telemetry will be available. Data on temperatures, thermostat bilevels, and heaters along with limited information on the power subsystem will be provided through the MTV telemetry system via the payload interface. This data is necessary to verify the health of the satellites prior to separation.

Prior to separation, each payload's TT&C subsystem will be activated to provide data directly to ground controllers and Earth. The attitude control subsystems must also be turned on and spacecraft batteries will be fully charged. After separation, the satellites will operate autonomously from the MTV. There are basically two orbital insertion scenarios. The first to be used is the low inclination insertion used by the MaRCoS I and II vehicles. The second scenario is used by the MSM, probe dispenser, and rover-lander assemblies and is into a high inclination orbit. The difference between the two is the location of separation from the MTV.

3.8.1.1 Low-Inclination Orbit Insertion. The MTV will be in a hyperbolic orbit heading towards Mars where it will insert itself into a high-inclination orbit. The MaRCoS spacecraft will operate in a low-inclination orbit. It is well known that fuel requirements for large inclination changes decreases with altitude. Therefore, the most efficient means of inserting a low inclination satellite will be to initiate the maneuver many planetary radii from Mars. The sequence of placing the MaRCoS vehicles into low-inclination orbits is described next.

At a specified distance from Mars, the MTV will separate from each of the MaRCoS payloads. This is accomplished using some combination of explosive bolts, springs, and cold gas thrusters on board the MTV. After separation, the satellite and its insertion motor will spin-up for stabilization, reorient itself along the correct thrust vector, and damp out all nutations in the angular momentum vector. Once orientated properly, a high-thrust burn is used to vector into a low inclination orbit around Mars. These maneuvers must be very precise and must correct any errors caused by the separation process; therefore, they must be performed by a liquid propulsion unit of some sort as solid propellants cannot be controlled accurately enough. As a result of the separation process, the MaRCoS spacecraft will be on a hyperbolic orbit at zero degrees inclination with respect to Mars. The exact distance at which this operation takes place depends on a number of parameters not yet available. As the exact transfer orbit is determined, this detail can be specified more precisely.

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[22] Tables 11 and 12 give a synopsis of spacecraft design.

When the payload approaches an altitude of 17,070 km above Mars, the orbit insertion subsystem will ignite to circularize the orbit at synchronous altitude. This requires a total change in velocity (ΔV)[23] of about 1,900 m/sec. The final ΔV can be performed by either a solid or liquid propulsion unit; however, in lieu of liquid propellants being used for the separation sequence, the basic hardware is already available for the final sequence burn as a liquid propellant. It is therefore, more economical to maintain the liquid motor subsystem for the final orbit insertion sequence and add additional consumables. The choice of propellant is based on a propellant's specific impulse (Isp). Of the available propellants, bipropellants (such as nitrogen tetroxide-monomethylhydrazine) have the highest specific impulse and are well understood. Equation 10 describes the required bipropellant mass (1:164) for the change from the hyperbolic transfer orbit to a circular orbit.

$$m_p = m_{bo}(e^{\Delta V/I_{sp}g} - 1) \tag{10}$$

where:

m_p = orbit insertion propellant mass (kg)

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[23] ΔV is the change, or delta, in velocity required to transfer from one orbit to another.

$$m_{bo} = \text{final mass of the satellite after burnout (kg)}$$
$$\Delta V = 1900\ m/sec$$
$$I_{sp} = \text{propellant specific impulse} = 300\ sec\ (1:166)$$
$$g = 9.8\ m/sec^2$$

Therefore, m_p is approximately 890 kg. We must add to this the propellant required for spin-up, reorientation, and spin-down maneuvers which this is approximately 4.5 percent more propellant (1:51). Finally, propellant plumbing, thrusters, valves, and pumps must also be included. These items are approximated as 8.4 percent of the total propellant mass (1:46). The resulting orbit insertion subsystem mass is 930 kg of bipropellant and 80 kg of propulsion hardware. After the final insertion burn, the payload will spin down and eject the orbit insertion subsystem.

3.8.1.2 High-Inclination Orbit Insertion. The orbit insertion subsystem for the MSM, probe dispenser, and rover-lander assemblies are similar to that for the MaRCoS. However, the MTV will provide most of the energy to insert the vehicles into a high altitude, circular orbit at 92.7 degrees inclination. After separation, the payloads will spin-up, reorient themselves, and damp-out nutations. The separation phase will initiate a Hohmann transfer to a 360 km orbit. Following a half-period travel time, the orbit insertion engine will fire near periapsis. Similar to the low-inclination orbit insertion subsystem, a ΔV of approximately 1,900 m/sec is required to enter a 360 km altitude, sun-synchronous orbit from a high Martian orbit.

Because this use of common hardware will simplify the integration effort required and reduce cost for this phase of Project Ares, the orbit insertion subsystems will be identical in design. The only difference among the payloads will be the amount of propellant used for insertion. The MSM and rover-lander assemblies are expected to weigh approximately the same as a MaRCoS vehicle. The probe dispenser is expected to be nearly twice that of the MaRCoS vehicle. Two alternatives are available. The first is to build a slightly larger subsystem based on the dispenser/probes' weight. The second is to break the dispenser into two vehicles carrying half the probes each. This decision can be made using other factors as either choice can be reasonably made.

3.8.2 Orbital Operations. Having been properly placed into their operating orbits, each spacecraft will require support for TT&C, attitude control, electrical power generation and distribution, and thermal control. This section describes the designs for each of these systems. Each subsystem design is based on various satellite requirements and trade-off analyses. These are discussed individually in this section.

3.8.2.1 Telemetry, Tracking, and Commanding (TT&C). This subsystem will provide transmission of spacecraft telemetry, and the reception, decoding and routing of commands within the vehicle. The equipment includes antennas, transponders, telemetry units and a single command decoder.

Telemetry is transmitted and commands are received through low-gain antennas located on the forward and aft faces of the spacecraft. The antennas provide near global coverage for nominal and contingency operations (37:2-22) which gives the system the ability to operate even in the event of loss of attitude. In such a case, this configuration will facilitate recovery of the vehicle without lose of commanding capability. Initially, both telemetry and commanding were to be combined with mission data, but, mission data is transmitted over highly directional beams and would be severely degraded in the event of a loss of attitude control.

Commands will be received by the transponder and routed to the command decoder where they are processed. The commands are simple binary structures, 0 and 1, representing single commands or data streams. Commands will be checked for correct structure, vehicle command count and parity and will be routed to the correct subsystem for execution (37:2-29). For large attitude control software loads, the command decoder can send echos of the received command for ground verification. The attitude control system can also dump the contents of the random access memory (RAM) onto the data stream in place of normal telemetry for further verification (37:5-38).

The telemetry unit will receive analog, serial digital and bi-level data from all subsystems. Telemetry will consist of equipment status, temperatures, currents, voltages, power usage, and attitude data from the control electronics. Multiplexers and analog/digital converters and the timing unit will assemble the telemetry data stream and send it to the transponder where it will be modulated onto the transmitted signal. Telemetry will be sent at a data rate of nearly 1 Kbps (37:5-8).

3.8.2.2 Attitude Control. Attitude control for most artificial satellites is accomplished in two different fashions: spin-stabilized and three-axis stabilized. Spin-stabilized satellites use basic gyroscopic principles to maintain pointing. The main body of the spacecraft is spinning at a fixed rotational velocity with the sensors or antenna on a despun platform pointing toward Earth (23:177). Spin-stabilization is mainly used for satellites which do not require high pointing accuracy (that is those which have a wide FOV).

Three-axis stabilized satellites use sensors, reaction/momentum wheels, and thrusters to maintain accurate pointing in pitch, roll and yaw. In this control mode, the actuators and sensors attempt to maintain pointing by keeping the motion of the satellite to a minimum. Three-axis stabilization provides the most accurate pointing of all methods. Due to the pointing requirements of the MaRCoS and MSM satellite, the logical choice for stabilization would be three-axis, nadir pointing, and zero momentum controlled.

The probe dispenser could use either spin or three-axis stabilization. If the dispenser was to remain in orbit following insertion of the probes, to provide redundant communication capabilities, then a three-axis stabilized platform is preferred.

The attitude control system (ACS) is comprised of three major hardware categories: actuators, sensors and electronics.

Actuators consist of four reaction/momentum wheels and thrusters. The reaction wheels are mounted 45 degrees to the pitch axis. All four wheels maintain pitch attitude while two of the four maintain roll and two maintain yaw. The reaction wheels control the attitude for an axis by storing or unloading momentum as required (23:206). The attitude control electronics send drive signals to the wheels in response to sensed errors in attitude. Only three wheels are needed to maintain control in pitch, roll and yaw (37:3-264); however, the arrangement of the four reaction wheels in this configuration provides redundancy in case of failure. Thrusters and the propulsion system will provide spacecraft attitude adjust capability and momentum wheel dumping. The wheel's momentum can be unloaded daily using the thruster's automatic control provided by the attitude control electronics. Since there is no magnetic field on Mars, thrusters were chosen for unloading momentum from the reaction wheels, in lieu of using magnetic torquors (89:161).

Current sensor technology is sufficient for the design of Mars orbiting spacecraft. Using the DSCS III as a baseline, the following methods of control and sensor fusion are possible.

Primary control of pitch and roll axes will be maintained by a circular IR sensor consisting of eight elements. The sensors main task is to report the position of Mars, which will be centered within the array elements. If the planet moves into the sensor's FOV, equating to a pitch or roll error, a signal is generated and sent to the electronics to correct the error(37:3-247). To prevent any unwanted signals from interfering with the sensor, a ring of eight silicon detectors surround the IR detectors. These detectors work in the visible range and will sense the approach of the sun. This sun sensor will warn the electronics that the sun will impinge on the IR sensor. The electronics will shut off elements of the IR detector to prevent the generation of false signals (37:3-249).

Yaw axis control will be maintained by two systems. Control within two hours of noon and midnight local satellite time (LST) will be maintained by rate gyros. These sensors will detect movement or rates in the yaw axis. The electronics will use these signals to null the errors as it does with signals from the other sensors. For control of the spacecraft at six am and pm LST, sun sensors mounted on the solar arrays will detect yaw attitude errors (37:3-261). These sensors will not detect errors but sunlight intensity. One sensor will sense a larger solar intensity if an attitude error occurs. Due to the tilt of the planet's axis, a sun declination bias will be calculated daily to compensate for differences in the signals due to the relative declination of the sun to the satellite (37:3-160). As a backup role, the sun sensors can be used to maintain pitch and roll control (37:3-261).

Tachometer data from the reaction wheels and data from the sensors are fed into the attitude control electronics. The computer electronics use algorithms to calculate the current position and motion of the satellite and outputs signals to the actuators to correct for any errors in attitude. This computer will use the data from the sensors, stored star data and ground ephemeris to maintain nadir pointing (70:5). All functions of the ACS will be controlled from embedded software found in on-board programmable read-only memory (PROM) with alterations to the programs uplinked and stored in RAM and accessed via breakpoints in the embedded software. This will allow for correction of deficiencies in software or hardware. The ACS will maintain pointing of ± 3 mrad pitch, roll and yaw (70:4). The ACS will also provide stabilization and thruster control for both inclination and velocity change maneuvers.

Because the low Mars orbiting vehicles have smaller tolerances for error, their sensors differ slightly from those used on the MaRCoS vehicles. Sensors consist of rate gyros, accelerometers, horizon sensor, 4π steradian sun sensor and a star tracker (70:4). One fixed star tracker and the horizon sensor will provide measurements for pitch and row axis control. Four rate gyros and the accelerometers will provide rate sensing for yaw control (75:263). Unlike the star trackers, the rate gyros do not sense position, only rate of motion in associated direction. The 4π steradian sun sensor will provide a reference after separation and for anomaly resolution (70:4).

3.8.2.3 Propulsion and Propellant. After the orbit insertion subsystem completes the final burn and the subsystem is ejected, the satellite must correct for any errors in altitude or inclination and then move towards the desired Martian longitude using the propulsion subsystem. The propulsion systems differ slightly for operations at low and high altitudes.

Synchronous Operations. After the satellite is operational in a synchronous orbit, a variety of perturbing forces will act on the satellite, causing the orbit to shift slightly. The major sources of perturbations are the sun, and Mars gravity potential variations. The propulsion system must correct any errors and maintain the desired synchronous orbit for ten Earth years. Using geosynchronous satellites as a baseline, the expected ΔV requirements in Mars synchronous orbit are 10.7 m/sec ΔV maneuver every 86.14 days for inclination corrections and 0.15 m/sec ΔV maneuver every 31 days for longitude corrections (1:88,91). For a 10 Earth year operational life, the total ΔV is 471 m/sec. These figures are very conservative—for a real mission, actual ΔV requirements must be calculated.

Propulsion subsystem options include monopropellants, resistojets, bipropellants, and ion thrusters. The first three have specific impulses near 280 seconds while ion thrusters have a specific impulse of 3,000 seconds. This directly impacts the subsystem mass according to Equation 11 (1:176).

$$m_p = m_{sat}(e^{\Delta V/\eta I_{sp} g} - 1) \tag{11}$$

where:

m_p = propulsion subsystem propellant mass (kg)

m_{sat} = satellite mass = 900 kg

$$
\begin{aligned}
\Delta V &= 471\ m/sec \\
\eta &= \text{thruster efficiency} = 0.844 \\
I_{sp} &= \text{propellant specific impulse} = 3000\ sec(1:166) \\
g &= 9.8\ m/sec^2
\end{aligned}
$$

Additional propellant mass must be included to account for station repositioning, attitude control, pressurant, and margin— this adds approximately 18 percent to the propellant mass (1:51). The mass of propellant plumbing, thrusters, valves, and pumps must also be included. Using a hydrazine system as a baseline, these items are approximately 4.6 percent of the total satellite mass (1:45). The resulting propulsion subsystem mass is 20 kg of xenon propellant and 40 kg of propulsion hardware. The ion thrusters will require 300 watts of electrical power.

Low Orbit Operations. The ΔV requirements for orbit correction, station keeping, and attitude control in low-Martian orbit will be higher than for the synchronous orbits. As a rough order of magnitude, the low Mars orbit vehicle's ΔV requirements are double that of the MaRCoS which are conservative to begin with. For a real mission, the actual ΔV requirements must be calculated.

Since the MaRCoS is using ion thrusters for station-keeping, the MSM, dispenser, and rover-lander assemblies will use them too. This use of common hardware will simplify satellite design and reduce cost. As a margin of safety, the ion thrusters on the MSM will be twice as powerful as those for the communications satellite. Ion thrusters generate 20 mN of thrust and require 600 watts of electrical power. The resulting propulsion subsystem mass is 40 kg of xenon propellant and 80 kg of propulsion hardware.

3.8.2.4 Electric Power Generation and Distribution. Power must be generated by either solar or nuclear power sources. The mass of a solar power system depends on many factors, such as solar cell type and operating temperature, solar

Table 8. Table on Battery Energy Density and Cycle Life (23:124).

Cell Type	Energy Density W-hr/kg	Cycle Life at 25 percent	Cycle Life at 50 percent	Cycle life at 75 percent
Ni-Cd	25 to 30	21,000	3,000	800
Ni-H_2	50 to 80	>15,000	>10,000	>4,000
Ag-Zn	120 to 130	3,500	750	100

panel design, battery type and depth of discharge, power conditioning architecture and efficiency, and others. The mass of a nuclear power system also depends on many factors, such as thermal output, static or dynamic conversion efficiency, heat rejection temperature, radiator design, shielding requirements, power conditioning architecture and efficiency, and others. Since the transportation cost to Mars is very expensive, the primary means to compare solar and nuclear power systems is on a mass basis.

For solar power systems, the three main components are the battery, solar array, and power conditioning system. To provide for a modular design, each unit is sized for 450 watt blocks. The MaRCoS requires 1,800 watts, thus four blocks are needed. For the Mars synchronous orbit, approximately 1,350 eclipses will occur over the satellite life. During these eclipses, energy storage cells must provide for all electrical power requirements. Nickel-hydrogen batteries operating at 80 percent depth-of-discharge can meet the required cycle life with minimum mass (23:124). A single string of 26 cells (30 amp-hour capacity, each cell 0.5 kg) can supply 450 watts of electrical power for the maximum 79 minute Martian eclipse time.

The cell string can tolerate a single cell failure: cell bypass diodes allow the string to continue functioning. For redundancy, one extra cell string will be carried on the satellite (1:372, 375).

Nickel-cadmium batteries also meet the cycle life requirements, but are at least twice the mass of nickel-hydrogen. Silver-zinc batteries meet the cycle life requirements, but the 30 percent depth-of-discharge limit offsets the high energy density. The resulting battery mass is still 50 percent heavier than nickel-hydrogen (23:124, 126).

For the solar array, power must be generated to supply the payloads directly and to recharge the batteries. The solar constant at Mars determines the solar array output: 708 W/m^2 at perihelion and 488 W/m^2 at aphelion. To meet the worst case design, the solar array must provide the required power at aphelion. To minimize solar array mass, the solar cells must have a high conversion efficiency: this favors gallium arsenide over silicon. Radiation damage effects must also be minimized. This also favors gallium arsenide cells. Solar array temperature also influences efficiency. By assuming constant coefficients of absorption and emission, the equilibrium temperature at aphelion is calculated as 240 K. Gallium arsenide efficiencies can reach 20 percent at 300 K, but since the cell is operating at a lower temperature, the efficiency is increased to 23 percent. To provide 450 watts of electrical power to the payloads, the solar array must be able to generate 600 watts End-of-Life (EOL) (23:107, 166). Slip rings are required to transfer electrical power from the solar array to the satellite bus. The MaRCoS slip ring design will be no more complicated than existing designs in use on board geosynchronous satellites.

By including the efficiency, assembly, environmental, temperature, and geometric packing effects from table 9, a 7.5 m^2 area is required for each 450 watt unit. Since minimum mass is required, a 3 kg/m^2 flexible blanket design (similar to the Space Station Freedom design) is the only choice (1:285, 342, 370, 376, 377).

The last major power subsystem component is power conditioning. These devices are very mission specific, so the mass and efficiency estimates will always be very simple. The basic architecture will be based on a direct energy transfer (23:148). Power conditioning equipment to control the bus voltage and amperage masses about 10 kg/kW. The wiring harness to carry electrical power from the bus

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Table 9. Solar Array Efficiency Factors (1:377).

Factor	Approximate Effect
Solar cell efficiency at operating temperature	0.23
Solar cell geometric packing factor	0.90
Assembly Losses	0.96
- Module assembly	
- Cell voltage mismatch	
- Measurement errors	
- Solar panel wiring loss	
- Block diode drop	
- Array wiring harness and slip rings	
Environmental Degradation	0.81
- Micrometeoroids	
- Ultraviolet light degradation	
- Adhesive and coverglass darkening	
- Low Energy Protons	
- Radiation	
- Temperature cycling and random failures	

to each payload masses about 20 kg/kW. The total power conditioning subsystem efficiency runs about 90 percent (1:17, 39).

For a 450 watt EOL solar power system, the battery string will mass 13 kg, the solar array will mass 22 kg, and the power conditioning hardware will mass 5 kg. The total mass for 1,800 watts is 160 kg, plus an additional 13 kg for the redundant battery string and 7 kg for margin. The wiring harness masses 36 kg plus 4 kg margin.

For nuclear power systems operating near 1,800 watts, the Dynamic Isotope Power System (DIPS) provides the lowest mass. DIPS uses a subcritical mass of plutonium as the heat source driving a Brayton power cycle. DIPS provides up to 2,000 watts for seven years with a total mass of 215 kg, including the radiation shield. This is heavier than the equivalent solar power system mass of 180 kg. DIPS would also have to be modified to meet the ten year life requirement. The biggest disadvantage is the Brayton conversion unit. The constant spinning of the turbine blades may introduce very small but constant jitter into the satellite. This would be detrimental to the very tight pointing and tracking requirements of the Earth transmission antenna. The only other space nuclear power alternative uses a critical reactor core and out-of-core thermionic cells. However, these systems will be heavier than DIPS at the 1,800 watt power level. With these factors in mind, the solar power system is chosen for the MaRCoS.

The MSM power subsystem can be either solar or nuclear. The requirements are 1,350 watts at EOL with minimum mass. Using the modular solar power subsystem design from the MaRCoS, the MSM will need three 450 watt blocks. Since the satellite will be in a sun-synchronous orbit, the eclipse power requirements are negligible; however, the satellite will experience eclipses during the orbit insertion maneuvers, and an energy storage system will be required during normal operations as an emergency backup. Using common hardware, the MSM will use a string of 26 nickel-hydrogen 30-amp-hour cells for each 450 watt block of electrical power. Since the satellite is in a low orbit, the charge current will be doubled— this will reduce the required recharge time.

The MSM solar array will also use gallium arsenide solar cells, but the operating parameters will be different from the MaRCoS. Since the satellite is in low orbit, reflected sunlight and thermal emissions from the Martian surface will contribute to higher operating temperatures. By assuming constant coefficients of absorption and emission, the solar array equilibrium temperature at noon and aphelion is calculated as 268 K. Since the gallium arsenide cell is operating at a low temperature, the efficiency is increased to 21 percent. To provide 450 watts of electrical power to the payloads, recharge the batteries, and provide a ten percent margin, the solar array must be able to generate 650 watts at EOL. An 8.7 m^2 area is required to do this. The flexible blanket solar array design masses 3 kg/m^2.

Power conditioning equipment for the MSM will also be very mission specific. The exact voltages, amperages, transmission distances, and electrical interference effects are unknown. For simplicity, the same estimates as for the MaRCoS will be used: 10 kg/kW for power conditioning, 20 kg/kW for the wiring harness, and 90 percent efficiency.

For a 450 W EOL solar power system, the battery string will mass 13 kg, the solar array will mass 26 kg, and the power conditioning hardware will mass 5 kg. The total mass for 1,350 watts is 132 kg, plus 13 kg for the redundant battery string and 5 kg for margin. The wiring harness masses 27 kg plus 3 kg margin.

For nuclear power systems operating near 1,350 watts, DIPS is still the lowest mass option. The DIPS mass of 215 kg still does not compare favorably against the equivalent solar power system mass of 150 kg. For this reason, the solar power system is chosen for the MSM.

The rover-lander assemblies have already been discussed. The probe dispenser's operational capabilities are still not totally known. Therefore, decisions regarding this system must wait for now.

3.8.2.5 Thermal Control. As with earlier systems, the operation of the thermal control subsystem depends on the thermal conditions of the chosen orbit. We first address the synchronous orbiting platform and then the low orbit platforms.

Synchronous Thermal Subsystems. Table 10 gives high and low temperature restrictions for subsystems on-board a typical geosynchronous satellite. These limits will also apply for satellites in a Mars orbit.

Many components in the MaRCoS will only operate within very narrow temperature limits. For a solar power satellite, the major source of heat comes from absorption of solar energy. Other sources include reflected sunlight and IR emissions from the Martian surface and internal dissipation of electrical energy due to inefficient electronics. The heat rejection capacity is determined by a number of factors, including satellite absorption and emission coefficients, radiator panel size, and the physical satellite geometry. View factors from one part of the satellite to another part can make or break the thermal subsystem design; however, this requires a detailed knowledge of satellite geometry, surface properties, and internal heat transfer characteristics. These parameters are input to a computerized thermal analysis code to develop an actual thermal subsystem design. Since these parameters are not available, a bulk satellite temperature response is calculated based on a spherical, uniform, perfectly conductive satellite (assuming radius = 1.25 m). The resulting satellite heat balance is described in Equation 12.

$$\epsilon_s \sigma T_s^4 A_t = \alpha_{vis} E A_p + \alpha_{vis} \rho E A_p' (\Omega/\pi) + a_{IR} \sigma T_M^4 A_p' (\Omega/\pi) + \Phi \qquad (12)$$

Table 10. Thermal Design Temperature Limits (degrees C) (1:266).

Equipment	Non-Operating Min / Max	Operating Min / Max
Communications		
- Receiver	-30 / +55	+10 / +45
- Input multiplex	-30 / +55	-10 / +30
- Output multiplex	-30 / +55	-10 / +40
- TWTA	-30 / +55	-10 / +55
- Antenna	-170 / +40	-170 / +90
Electric Power		
- Solar array wing	-160 / +80	-160 / +80
- Battery	-10 / +25	0 / +25
- Shunt assembly	-45 / +65	-45 / +65
Attitude control		
- Planet/sun sensor	-30 / +55	-30 / +50
- Angular rate assembly	-30 / +55	+1 / +55
- Momentum wheel	-15 / +55	+1 / +45
Propulsion		
- Propellant tank	+10 / +50	+10 / +50
- Thruster catalyst bed	+10 /+120	+10 /+120
Structure		
- Pyrotechnic mechanism	-170 / +55	-115 / +55
- Separation clamp	-40 / +40	-15 / +40

where:

ϵ_s = IR emission coefficient for the satellite

σ = Boltzman's constant = $5.67\ x\ 10^{-8}$

T_s = satellite temperature

A_t = total surface area of the satellite = $19.6\ m^2$

α_{vis} = visible absorption coefficient for the satellite

E = solar constant

$708\ W/m^2$ at perihelion,

$488\ W/m^2$ at aphelion, and

$0\ W/m^2$ in eclipse

A_p = perpendicular area of the satellite towards the sun = $4.9\ m^2$

ρ = Mars visual albedo = 0.15

A'_p = perpendicular area of the satellite towards Mars = A_p

Ω = solid angle of Mars as seen by the satellite = 0.087 *steradians*

a_{IR} = IR absorption coefficient for the satellite = ϵ_s

T_M = Mars blackbody radiation temperature

$227°K$ at perihelion and

$207°K$ at aphelion

Φ = internal thermal power dissipated by the satellite

1260 *watts* assuming 1800 W at 30 percent efficiency

The coldest temperature will occur at midnight and aphelion. Assuming the internal thermal power dissipated is 70 percent of the 1,800 watts total electrical power, and assuming the minimum satellite temperature is 273 K, the IR emission coefficient for the satellite must be less than 0.204. The hottest temperature will occur at noon and perihelion. Assuming the internal power dissipated is 70 percent of 1,800 watts, maximum satellite temperature is 313 K, and an IR emissivity of 0.2, the visible absorption coefficient for the satellite must be less than 0.258. The coefficient limits can be approximately met by using a flat reflective surface, such as aluminum paint (95:1577).

A real thermal control subsystem design would have to consider many other factors. The satellite is not spherical, uniform, and perfectly conducting. However, Equation 12 does say that the satellite will run cold most of the time. Therefore, most of the satellite surface will be insulated, and sensitive items (such as the propulsion system and batteries) will have electrical heaters. For the MaRCoS, a heater power of 100 watts is included in the power subsystem requirements. Only a few small radiator panels are required. These panels will have high IR emissivity and will typically be mounted on the satellite north side (facing up, away from the Sun). The exact panel characteristics must be determined by calculation and by thermal testing of actual satellite hardware. Environmental degradation effects must also be included. As a first approximation, the mass of the thermal control system is given by Equation 13 (1:49).

$$m_t = 0.04\ \Phi \tag{13}$$

where:

m_t = thermal control subsystem mass (kg)

Φ = internal thermal power dissipated by the satellite 1260 *watts* (1800 W at 30 percent electrical efficiency)

This results in a thermal control system mass of about 50 kilograms.

Low Martian Orbit Thermal Subsystems. Like the synchronous orbit conditions, many of the detailed thermal control subsystem design parameters are unknown for the low Mars orbits. As a first estimate, a bulk satellite temperature response is calculated based on a spherical, uniform, and perfectly conducting satellite (assuming radius = 1.25 m). The MSM thermal balance is described in Equation 12 with the following variable changes: the solid angle of Mars as seen from the satellite is 3.56 steradians and the internal power dissipated is 70 percent of 1,350 watts. To maintain a minimum satellite temperature of 273 K at midnight and aphelion (which may occur during the orbit insertion maneuvers), the IR emission coefficient must be less than 0.173. To maintain a maximum satellite temperature of 313 K at noon and perihelion (which is conservative since the satellite will never reach Martian noon), the visible absorption coefficient must be less than 0.171. The coefficient limits can be met by using a flat reflective surface; however, the existing aluminum paints will not be sufficient (95:1577). These results say that the MSM will operate warmer than the MaRCoS, but will still tend to run cold most of the time. One hundred watts of heater power is included in the power system design for sensitive items, such as the propulsion system and batteries. For the real satellite, a few radiator panels mounted on the satellite's north side will handle the required thermal loads. This same design can be extended to the probe dispenser design.

3.8.2.6 Structures. The structural mass of a satellite is very mission dependent. The most stressful loads are imposed during launch from the Earth's surface and during orbit insertion maneuvers. These loads include launch acceleration, acoustic noise, impulse shocks, random accelerations, spin stabilization, and vibrations. The satellite structure must be designed to hold the various payloads and subsystems together during these mission phases. The structural mass required to hold an antenna together can often be almost as much as the mass of the antenna itself. For a real satellite design, very detailed simulations and tests must be run using the actual satellite geometry, component masses, and structural characteristics. Since this information is not available, an empirical equation must be used to estimate the structural mass.

Synchronous Vehicles. For geosynchronous satellites, the structural mass is approximately 8.7 percent of the separation mass (1:48). The satellite separation mass includes everything the structure must support during launch and orbit insertion operations. This assumes the structure is made mainly out of aluminum. By using advanced composite materials such as metal matrix, graphite/epoxy, or thermoplastic materials, the structural mass can be reduced to 6.1 percent of the separation mass (1:48). Using a separation mass of 1,910 kg, the mass of a composite satellite structure is estimated as 115 kg plus a 5 kg margin.

Low Martian Platforms. The MSM should be shaped similar to a Satcom-K based structure with a central body which is capable of carrying the solar array, high gain antenna, and payload (70:2). The main body will be 1.5 by 1.5 by 2.1 meters (62:81).

The primary structure will support the optical instruments and electronics, avionics package, as well as the data handling element. Particular attention must paid to the location of the disk storage devices of the data system to minimize the gyroscopic and momentum effects of these units. Mounted on the structure will be the secondary instruments that will measure magnetic field strength and primary and secondary gamma rays among other experiments (70:7). The primary structure will also provide mounting interfaces to the MTV. This will include any devices required for separation after reaching the Mars vicinity and wiring harnesses required for sharing of power and TT&C with the MTV.

The solar array and the high-gain antenna will be mounted to the main body via canister boom systems (70:7). Deployment of the high-gain antenna, solar array, and select secondary instruments will be accomplished by mechanisms consisting of stepper motors, spring hinges, and pyrotechnic devices (70:8).

The high launch and orbit insertion loads expected for MaRCoS will also apply directly to the MSM. Since the satellite separation masses are identical at 1,910 kg and since the same structural loads are expected, the same structural mass is required. Therefore, the mass of a composite structure for the MSM is estimated as 115 kg plus 5 kg margin.

3.8.3 Summary. The design of Phase II mission platforms involved many considerations based on mission environment and objectives. This section laid the groundwork for further refinement of exact design parameters. We have given an estimate of the final platform design weights and specifications but these are only estimates and serve only to guide future mission designs. A summary of the MaRCoS and MSM spacecraft parameters is included in Tables 11– 12.

3.9 Mission Systems Summary

This chapter focused on the design of mission systems to provide data for future phases of Project Ares. We addressed overall objectives and requirements before delving into functional subsystems. Initially, we determined that there were two basic orbital locations. The first was a synchronous orbit at 20,424.67 km radius at zero inclination to be used by the two MaRCoS spacecraft placed approximately 170 degrees apart. The second orbit was at 360 km with an inclination 92.7 degrees which is a sun-synchronous location. The MSM would be placed in this orbit so that it would consistently observe the surface at 1400 hours. Next we looked into the communications system.

The design of the Mars communications system is dictated not only by the requirements of the various data collection elements in the Mars theater, but also by the interplanetary distances involved. Commands must be relayed to control the health and status, as well as the mission payloads, of the MaRCoS and MSM spacecraft and the probe/lander groups. Telemetry and mission data must then be

Table 11. MaRCoS Summary.

	Mass	Power	Comments
Separation mass and EOL power:	1910 kg	1800 W	
Orbit insertion propellant:	930 kg	-	bipropellant
Orbit insertion hardware:	80 kg	-	1900 m/s delta V
BOL mass and EOL power:	900 kg	1800 W	
Earth antenna:	100 kg	900 W	tight pointing
MSM data antenna:	TBD	TBD	pointing
MSM TT&C antenna:	TBD	TBD	omnidirectional
Mars surface antenna:	TBD	TBD	Mars hemisphere
Data storage:	TBD	200 W	spinning disks
TT&C:	30 kg	50 W	
Attitude control:	70 kg	50 W	
Propulsion:	40 kg	300 W	10 mN ion thrusters
Propellant:	20 kg	-	xenon
Electric Power:	180 kg	-	provides 1800 W
Distribution Harness:	40 kg	-	28 V bus
Thermal control:	50 kg	100 W	temp range 0-40 C
Structure:	120 kg	-	composites
Margin:	250 kg	200 W	

Table 12. MSM Summary.

	Mass	Power	Comments
Separation mass and EOL power:	1910 kg	1350 W	
Orbit insertion propellant:	930 kg	-	bipropellant
Orbit insertion hardware:	80 kg	-	1900 m/s delta V
BOL mass and EOL power:	900 kg	1350 W	
Sensors and mechanisms:	240 kg	150 W	optics
Data antenna:	TBD	TBD	pointing
TT&C antenna:	TBD	TBD	omnidirectional
Data storage:	TBD	200 W	spinning disks
TT&C:	30 kg	50 W	
Attitude control:	70 kg	50 W	
Propulsion:	80 kg	600 W	20 mN ion thrusters
Propellant:	40 kg	-	xenon
Electric Power:	150 kg	-	provides 1350 W
Distribution Harness:	30 kg	-	28 V bus
Thermal control:	40 kg	100 W	temp range 0-40 C
Structure:	120 kg	-	composites
Margin:	100 kg	200 W	

returned to Earth from those units and the twelve surface probes. All this must be accomplished as quickly as possible, but within the constraints of power budgets, data integrity standards, and realistic technology advances.

Frequency bands for each of the links involved (refer to Figure 7) were selected based on the amount of data required and environmental factors, but always with the goal of minimizing link-induced errors. For this reason, the Ka-band was chosen for high-data-rate links, while C-band is used for uplinks and communication with surface units. Nodes are also designed to provide complete redundancy whenever possible due to the catastophic consequences of single-point failures within a system located millions of kilometers from Earth. Multiple hardware strings, backup channels, a satellite crosslink, and contingency *workarounds* provide this redundancy. With millions of bits being handled across such extreme distances, data loss is also a primary concern. Both the MSM and MaRCoS have robust optical storage devices which minimize effects of occultations by storing large amounts of mission data for later transmission. Also, the convenience of a single uplink and downlink trunk via the prime MaRCoS requires autonomous and thoroughly reliable data routing. Having defined the support communication system for data gathering devices, we began analysis of the MSM.

The MSM is a near-polar orbiting surveyor with a five-year design life with the primary mission of landing site certification. Twelve potential landing sites will be examined by imaging the surface with a HIRES system. Because silicon CCDs are responsive over visible wavelengths and silicon technology is mature, the sensor uses silicon detectors with a linear *push broom* sweeping motion. The raw data rate from the HIRES sensor is on the order of 300 Mbps.

The MSM also has the added responsibility of mapping the entire Martian surface, at a medium resolution (MEDRES mission). This wide angle system consists of two lenses and one focal plane and has both a blue and a red bandpass filter. The expected data rate from this medium resolution system is on the order of 0.670 Mbps.

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The tertiary mission of the MSM is to conduct other scientific experiments (science mission). These experiments will be conducted by a tertiary mission payload complement consisting of six individual detectors producing, in aggregate, approximately 11,000 bps.

The next section examined the considerations governing the choice of a surface probe concept. These considerations included mission goals (site selection vs scientific exploration), specific mission objectives (meteorology/climatology, seismology, and surface/atmospheric chemistry), mission lifetime, power requirements and generation (RTG vs photovoltaic), delivery concept, deceleration techniques, landing options, and instrumentation. After examining the proposed designs of Burke et alia of the Jet Propulsion Laboratory, and weighing how well each satisfied the above constraints, we *propose* a modification of their soft lander or egg concept.

This proposed concept involves a total of 12 eggs, packaged in fixed aeroshells, which can be delivered on a single bus. The bus will have preset code to insert the eggs over 12 candidate sites selected by the MSM. The individual aeroshells provide initial deceleration, with subsequent braking supplied by parachutes, retropropulsion, and crushable coverings. Power is supplied by a small, lightweight RTG. Wherever possible, solid-state instrumentation is chosen over mechanical. Each egg has a required lifetime of one Martian year. Following insertion of probes and data gathering for approximately six months, the surface rovers will be deorbited.

The primary mission for this segment of the Project Ares is the gathering of data needed to make the final decisions regarding future manned missions. This data relates to hazards of manned missions in the Martian environment. A component-level analysis was performed in order to estimate the requirements levied on the other Phase II components (primarily MTV and MaRCoS) and to validate the final concept.

We determined that a mobile laboratory (rover), along with a landing and support platform (lander), could perform this mission. Operationally, there will

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be two rover-lander packages, one for each of the two sites deemed optimal from the data returned by the previous missions. The rover will evaluate the surface conditions in detail to ensure Mars is safe and suitable for man. The lander will safely convey the rover from orbit to the Martian surface, and also perform some scientific investigations. Both rovers will be active on Mars simultaneously. Communications will be direct to the Mars Relay Communication Satellites (MaRCoS) in stationary orbits with limited direct-to-Earth communications on a contingency basis.

The rover-lander pair will weigh just over 1,000 kg in orbit and 750 kg on the Martian surface. The rover will be a wheeled vehicle two meters long and one meter wide, not including wheels. The lander is a 2.5-by-2.5 m square (viewed from above) with four landing legs. Both the rover and the lander require less than 100 W of electrical power each. The rover lifetime will be one Earth year; the lander will last ten years.

Finally, we examined the supporting spacecraft. Although they are vital to the operation of this phase, the spacecraft were designed from standard engineering equations that have been well-understood and demonstrated over the last decade. This brings us to the last chapter of the report—design of an Interplanetary Transportation Vehicle (the MTV mentioned frequently throughout this chapter).

IV. Mars Transportation System

Project Ares is designed to place man on the Martian surface by the year 2014, with a buildup to an eventual permanent presence. The first phase of this five phase program is currently underway. Phase II begins with the return to Mars to gather scientific data of the Martian surface and environment prior to man's return. More specifically, Phase II puts in place the orbital and surface probes needed to gather this data in two missions—one in the year 2001 and the second in the year 2005. After the site selection process is completed, based on the data gathered in Phase II, Phase III initiates the placement of hardware on the Martian surface necessary to support the initial manned visit. This ambitious undertaking requires a system to transport the Phase II orbital and surface probes, as well as the increasingly greater amounts of hardware and supplies needed in the future phases to support man on Mars. This vehicle will be unlike any space vehicle in existence. The examination of the mission and system requirements of this MTV is the topic of this chapter.

The design of an interplanetary transportation system is a very complex undertaking. There are many variables, interrelationships, and trade-offs to consider. Ultimately, there is not a single optimal answer. The various mission requirements and subsystems of a complete interplanetary transportation system must fit together perfectly, like an intricate jigsaw puzzle—a puzzle, however, with many different ways, or options, to fit the pieces together.

The mission and subsystems requirements and options must be examined separately to analyze the support technologies and hardware necessary to develop the individual pieces. However, these pieces (mission requirements and subsystems) must be continuously integrated together to complete the design of the entire system. Throughout this process, an understanding of the trade–offs, interrelationships, and complexities involved in planning a mission of this size is realized. To effectively examine and understand these aspects, our analysis is divided into four major areas of concern and examined at a mission and subsystem requirements level. The following are these four major areas:

- *Specific Objectives and Requirements.* Examines the objectives and requirements of the transportation system with respect to the Phase II missions as well as the long term Project Ares objectives of supporting a permanent manned presence on Mars. This section also lists the critical assumptions made.
- *Mission Analysis.* Examines the orbital constraints and requirements, astrodynamic calculations, variations, and options available to spacecraft traveling to Mars. Representative ΔVs and TOF are given for the various orbital transfer trajectories available.
- *Mars Transfer Vehicle.* Examines the major subsystems of the MTV. An analysis of subsystem requirements and options is performed.
- *Support.* Examines the logistical requirements for the various vehicle design possibilities including Earth-to-orbit requirements and on-orbit support.

After an examination of the requirements and options available within these four major areas, a recommendation of a particular set of available options for a system that satisfies mission requirements is presented.

4.1 Specific Objectives and Requirements

Given the project to plan and examine hardware requirements necessary for the operation of Phase II of the five phase Project Ares program, the specific objectives of the transportation group are to:

- Research and analyze the mission and subsystem requirements for the development of a generic interplanetary transportation system capable of placing mission payloads into Mars orbit.

- Develop this system with respect to the support required by the overall program, Project Ares, not just that required to support the specific mission payloads of Phase II.
- Examine the interrelationships and trade-offs necessary in the design of a complex interplanetary transportation system.

The requirements of the Mars Transportation System are driven by two factors: specific Phase II mission payload requirements presented in Chapter 3 and more general requirements for future phases of Project Ares. These transportation system requirements are indicated below:

- *MTV with 6.5 mt Payload Capacity to Mars Orbit.* Maximum Phase II payload requirements occur on the first flight in 2001. The payload consists of two MaRCoS and one MSM with a total mass of 5.7 mt (reference Tables 11 and 12). With the addition of a 15 percent margin, the maximum payload requirement is 6.5 mt.
- *Concept Validation of Routine Transport to Mars.* The MTV used in Phase II will validate the concept of routine transportation to Mars. This requirement is driven by the necessity to provide routine delivery of the increased payload requirements of the future phases of Project Ares. Initially, this MTV will not be man-rated, but may be considered for such in future phases.
- *Earth-Mars Transit Time of Less Than 12 Months.* This requirement is again driven by Phase II payload requirements as discussed in Chapter 3. It is assumed that future phase payloads will not require transit time more restrictive than those in Phase II.
- *Round Trip TOF (Earth-Mars-Earth) Less Than 40 months.* The Phase II mission profile calls for one mission in 2001 and one mission in 2005 with Phase III missions beginning in 2009. As discussed in the Mission Analysis

section, Earth-Mars planetary alignment, or more importantly, similar transfer trajectory opportunities, repeat roughly every 26 months. Allowing a 12 month refurbishment period with 52 months between missions, drives the 40 month round trip requirement. This schedule allows a series of MTVs to provide routine (every 26 months) delivery of payloads in future phases to support Project Ares.

- *Simple and generic payload interface for the MTV.* Over its lifetime, the MTV will carry various types of payloads in support of Project Ares. A simple and generic interface is required to simplify the training and operations required to support the mating of the payload to the MTV.

Three major assumptions of the Transportation Group which were previously identified in Chapter 1, are critical to the development of an interplanetary transportation system and are restated for completeness. They are:

- *Space Station Freedom Availability.* It is assumed that SSF is fully operational and has been expanded to support the assembly, test, payload mating, launch, maintenance, refurbishment, and refueling of the MTV.
- *National Launch System Availability.* It is assumed that the NLS is fully operational and has a maximum payload capacity to LEO of 250 mt.
- *Space Transportation System Availability.* It is assumed that the STS, or a follow-on vehicle capable of transporting men into LEO, is available.

4.2 Mission Analysis

Mission Analysis is a critical part of the design of an interplanetary transportation system. In dealing with interplanetary travel, manuvering in space is not just a simple matter of aiming your spacecraft toward a desired destination and firing the rockets. Orbital mechanics is a very complex science with many variables and no simple answers. While there are an infinite number of combinations of paths or trajectories from Earth to Mars, there may be only a few that are feasible given mission requirements and vehicle constraints.

To examine the options available for travel from Earth to Mars, a review of general orbital concepts is performed. Types of transfer trajectories and applicable variations are examined followed by a first order analysis of the orbital mechanics needed to understand the complexities involved in calculating the specifics of the orbital path. Finally, representative transfer trajectory specifics, including ΔVs and TOF requirements, are presented.

4.2.1 Mission Trajectories. The first consideration in mission design is the particular trajectory the vehicle follows in order to reach its destination. Mission performance and orbit selection are primarily affected by three factors: 1) the planetary alignment or geometry, 2) the transfer time between planets, and 3) the transfer mode chosen. The *geometry* involves solution of the particular two-body problem using basic orbital mechanics and applying perturbation theory techniques. The *transfer time* between the two planets is determined by the choice of a sprint trajectory or a low energy path. The specific combinations or *modes* dictate the overall mission design. An example of different modal types involve whether the vehicle travels on a direct or indirect path. A direct path means the angular transfer is less than 180 degrees while an indirect path implies greater than 180 degrees (16:86).

4.2.1.1 Background. There are two main paths or ballistic trajectories for traveling between Earth and Mars. They are the conjunction class (long duration) trajectory and the opposition class (short duration) trajectory. These two classes are also sometimes referred to as 1,000 day and 500 day voyages respectively (82:499).

Before exploring the specifics of the two types of classes, a few concepts must be explained. The angular positions of Earth and Mars vary cyclicly in two ways. First, the relative positioning of the two is the same approximately every 26 months (the synodic period) which means that their closest approach happens regularly at slightly greater than two year increments. Also, the same heliocentric positioning repeats on intervals of 15 years. This latter effect indicates that optimum launch times within that 15 year period will be repeated in subsequent periods. Therefore, calculations made for 1999–2014 will be good approximations for 2014–2029. It is the relative angular positioning of the planets that makes the largest contribution to minimizing the ΔV required for the interplanetary voyage (16:86).

4.2.1.2 Conjunction Class Trajectories. Conjunction class trajectories are characterized by a minimum energy (low thrust) transfer between the two planets on Hohmann transfer type trajectories[1]. Simplified calculations are performed assuming the two orbits are circular and coplanar. These missions have one-way trip times between 200 and 300 days accompanied by long periods remaining in the Martian system waiting for the proper planetary alignment for the return low-energy transfer back to Earth. This results in nearly equal trip times for both the outbound and inbound journeys. The transfer trajectory is characterized by a highly elliptical orbit that is tangential to the orbits of both Earth and Mars. The transfer orbit has a radius at periapsis equal to Earth's orbital radius and a radius at apoapsis equal to Mars' orbital radius(57:1–2).

Earth moves around the Sun at a faster rate than Mars, so Earth must be behind Mars relative to the Sun at the beginning of the transfer and will be ahead of Mars when the transfer vehicle arrives. This places the total travel angle near 180 degrees (reference Figure 19) (16:85).

Typical travel times between planets are approximately 260 days one way and are characterized by hyperbolic velocities [2] of 2.95 km/sec at Earth and 2.65 km/sec

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[1] A Mars conjunction is an event viewed from the Earth such that Mars and the Sun are located at the same celestial longitude (91:295).

[2] The hyperbolic velocity represents the excess energy in the orbital system when the potential energy is equivalent to zero. $v_\infty = \sqrt{2\mathcal{E}}$.

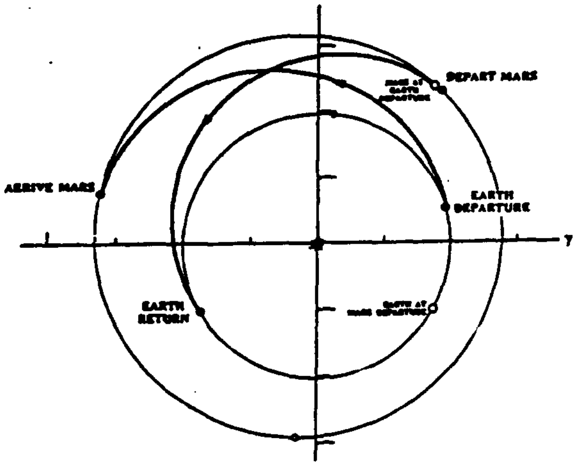

Figure 19. Conjunction Class Mission Profile (57:12).

at Mars. Because the time to wait for the correct planetary alignment is on a cycle of approximately 780 days, the stay in the Martian system would be 520 days. Adding the return travel time yields a total mission duration of 1,040 days (77:6).[3]

4.2.1.3 Opposition Class Trajectories. Opposition class trajectories involve higher energies. These missions occur when Earth and Mars are approximately lined up on the same side of the Sun. They involve the quickest trip times and possibly the shortest stays in the Martian system. These are high-thrust missions (16:85). Earth and Mars repeat closest approach once every synodic period (59:515–516).

Opposition class maneuvers are designed to reduce the overall mission duration by reducing the trip time in one or both directions. The two legs of the journey are not symmetric so only one leg is a low-energy transfer similar to that of the conjunction class mission (reference Figure 20). The other leg involves a high-energy burn which can occur on either the outbound leg or the inbound leg, depending on the particular planetary alignment at the desired launch time (77:6–7).

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[3] These are intended to be representative calculations only.

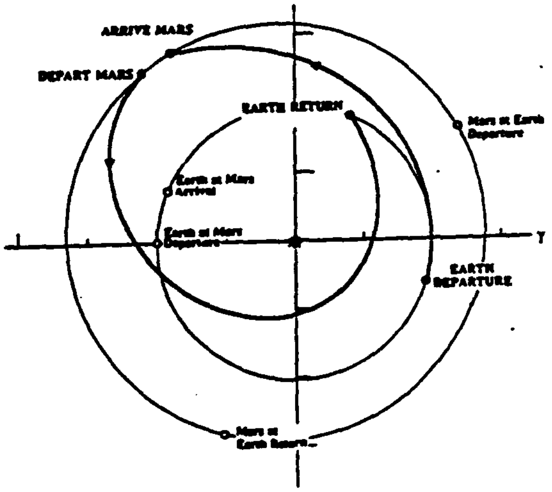

Figure 20. Opposition Class Mission Profile (57:12).

For opposition class missions, there are two possible methods: indirect/direct or direct/indirect. A direct path, as mentioned before, involves a Hohmann type transfer with an angular change of less than 180 degrees. The indirect path (angular change greater than 180 degrees) is unique, since its path crosses inside Earth's orbital radius to gain velocity to catch the quickly moving Earth. The direct/indirect combination has the overall ΔV advantage by leaving a large amount of mass in the Martian system. The indirect portion of the flight requiring the higher ΔV is then calculated with a smaller amount of mass (77:12).

The outgoing trajectory in a direct/indirect scenario is very similar to the conjunction class case. The main difference is the return trajectory. The stay at Mars is very small (30 to 60 days) and the path the vehicle follows on the inbound leg cuts inside of Earth's rotation to increase speed to catch the planet at an earlier time. The overall trip time is less than the time required to wait for the correct planetary alignment to occur before setting out on the return trip as in the conjunct ion class scenario. The characteristics for this transfer are short stay times in the Martian system and a huge difference in the inbound and outbound trip times. ΔVs are two to three times higher for opposition class missions due to higher velocities on either the inbound or outbound leg. The eccentricity of the Mars orbit has a greater effect on opposition-class missions(57:2–3).

4.2.1.4 Venus Swingby. Because the planetary alignment is not optimum for the high energy leg of the journey, the MTV's orbit actually passes closer to the Sun near the orbit of Venus (reference Figure 21). This results in opportunities to use a swingby maneuver around Venus to reduce the overall ΔV requirements of the mission. The relative positioning of Earth, Venus, and Mars repeats favorably every 6.4 years (the syzygistic period). Every five periods (the syzygistic cycle) the cycle repeats itself with two outbound and two inbound opportunities every period. There is usually at least one opportunity to use the Venus swingby for each mission profile (77:6–7).

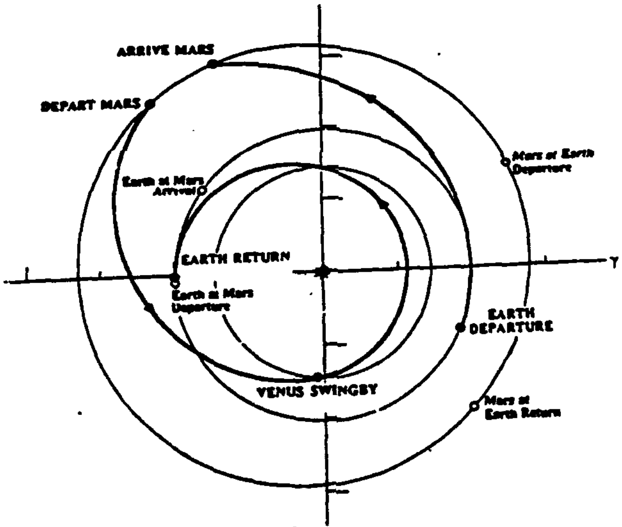
ARRIVE MARS
DEPART MARS
Mars at Earth Departure
Earth at Mars Arrival
EARTH RETURN
Earth at Mars Departure
EARTH DEPARTURE
VENUS SWINGBY
Mars at Earth Return
Υ

Figure 21. Opposition Class Mission Profile With Venus Swingby (57:13).

The primary reason for a Venus swingby is to remove energy from the trajectory so that the approach velocity is below tolerable limits. The trip time is reduced by moving within Earth's orbit, thereby increasing velocity. This velocity must be removed before entering into the desired orbit (77:15). Use of a Venus swingby on either the inbound or outbound trajectories reduces the ΔV requirement. A velocity assist maneuver around the planet reduces the propellant mass required, thereby reducing the initial mass needed in LEO. Not only does this improve the scenarios for reaching Mars with less mass, but it also allows the addition of several more launch opportunity times thus reducing the dependence on the opposition positioning of Earth and Mars (16:87).

4.2.1.5 Variations. Variations in the two-body, circular, coplanar solution are due to many effects. The eccentricity of Earth's orbit has minimal effects, while the eccentricity of Mar's orbit does produce noticeable variations that cycle on a 15 year period. Hohmann transfers are calculated as two-dimensional solutions that become much more complicated in three dimensions. The third dimension of the actual trajectory is due to the inclination difference of the Martian orbit compared to Earth's ecliptic.[4] For low thrust missions, the small inclination differences throughout the mission are accounted for by a series of small corrections; however, high thrust missions are penalized by greater requirements to enter the new plane. One possible solution is to perform a plane change maneuver at the intersection of the orbit of Mars and the Earth's ecliptic. This is the most efficient positioning to perform a small plane change maneuver (77:7–8).

Reducing the one-way flight time is possible for both classes of missions; however, the penalties are greater for the opposition class scenarios. Conjunction class missions could be reduced by 100 days (outbound and inbound) with only a five percent increase in the ΔV needed to escape from Earth. The round-trip duration remains the same for the correct planetary alignment, giving an increase in stay in the Martian system of 200 days. This particular scenario is appealing to manned missions, but the incoming ΔV to orbit Mars increases. This increase has to be accounted for either by increasing propellant mass or using aerobraking techniques. The same argument holds for the opposition class missions but there is a much larger increase in the arrival velocity (57:4).

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[4] Earth's ecliptic refers to its orbital plane around the Sun.

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4.2.1.6 Aerobraking and Its Complexities. By necessity, an interplanetary transfer orbit is a hyperbolic trajectory. Aerobraking is one possibility of transitoning from a hyperbolic trajectory to an elliptical orbit. Aerobraking uses air drag to provide a low-fuel method of reducing a satellite's kinetic energy. The Apollo program used aerobraking for the command module's return to Earth. When aerobraking is included in an interplanetary mission, a satellite must lose a specific amount of energy to achieve the desired orbit.[5] To lose the specified energy, the satellite must control the loss through aerodynamics. To design a mission using aerobraking, mission planners select the aerodynamics and transfer orbit characteristics and then attempt to find a flyable window for the satellite. Many parameters affect the amount of energy lost: entry speed, lift and drag characteristics, angle of attack, and atmospheric number densities at altitude (15:361).

Typically a satellite designed with aerobraking in mind will have some set of constraints placed on the flight envelope. For instance, heating rates and maximum acceleration limits are based on the technology available. Also, there are other constraints associated with the sensitivity to small variations in the entry conditions. For example, it is unreasonable to expect a satellite to enter the atmosphere within less than a degree of the desired angle of attack or to expect less than a one percent variation in the atmospheric density from a predicted result. There are also constraints dealing with the mechanics of the problem:

- *Altitude Limits.* If the altitude is too high, lift and drag disappear; too low, the satellite hits the ground.
- *Angle of Attack Constraints.* The under-shoot boundary is the point where the satellite has to execute a maximum lift up maneuver to avoid going too low. The overshoot boundary is the point where the satellite has to execute a maximum lift down maneuver to prevent going too high.

---

[5] Once most of the energy is lost, the satellite uses aerodynamic lift to pop up out of the atmosphere, and a ΔV at apoapsis raises periapsis out of the atmosphere.

- *Speed Limits.* If the speed is too fast, the satellite burns up; too slow, it fails to achieve the desired orbit.

The aerodynamics of the satellite are subject to many of the same limitations that relate to the flight envelope of an aircraft. The bigger the envelope, the easier it is to fly the satellite into the desired orbit (15:361– 262). Based on the constraints above, the following can be noted for a satellite with a lift to drag ratio of 1.0, and typical entry velocities ranging from 8.7–15.2 km/sec at Mars and 13.7–14.7 km/sec at Earth (67:590):[6]

- The typical deceleration times are about 100 seconds for both Earth and Mars.
- The heat-transfer rates for Mars and Earth aerobraking are 200–1500 W/cm^2 and 600–1100 W/cm^2, respectively.[7]
- The maximum g-loads are expected to be less than 6.5 g for Mars and 4.0 g for Earth.
- About 80 percent of our planned missions are achievable with aerobraking.

If a satellite has variable pitch lifting surfaces (wings), it could perform all currently planned missions. Additionally, the increased lift to drag ratio would: 1) increase the deceleration time by a factor of four, 2) decrease the heat transfer rates by a factor of six, and 3) lower the maximum g-loads by a factor of four. However, the increased performance is offset by the added complexities of control surfaces, on-orbit assembly due to unwieldy design, and the experimental nature of the technologies involved (86:516– 518).

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[6] A biconic has a lift-to-drag ratio of 1.0.

[7] Current technology is capable of producing unmanned systems capable of withstanding heat-transfer rates up to 2 kW/cm^2.

4.2.1.7 Launch Windows. Launch window variation affects the initial mass required for the mission. Chemical rockets are affected more than their nuclear counterparts. The penalty for a 30 day slip (± 15 days) in the optimum launch time results in a penalty of 9.6 percent of the initial mass for the chemical thrust vehicle. The penalty in weight for the nuclear thrust vehicle is only 6.6 percent, but this option tends to be more sensitive outside of a two-week window (77:17).

4.2.2 Orbital Analysis. The launch date fixes the respective positions of Earth and Mars at that instant allowing for the general two-body solution using Kepler's laws and patched conic approximations (patching elliptical orbits at both planets with a hyperbolic trajectory in between). This is a first order approximation to the actual trajectory of the spacecraft. Using perturbation analysis techniques, actual ΔVs for mid-course correction could then be computed to arrive at final boundary value conditions. Once the vehicle leaves Earth's sphere of influence (9.29×10^5 km (52:289)), the vehicle basically follows a two-body trajectory with the Sun as the main attracting body (68:427).

4.2.2.1 Patched-Conic Method (52:83–102) (12:357–384). The patched conic method assumes all gravitational perturbations from non-primary objects are zero. Assume a spacecraft in Earth vicinity experiences the acceleration of the Earth, but not the Sun. After leaving Earth orbit, the spacecraft is assumed to be under the gravitational acceleration of only the Sun. Once in the vicinity of Mars, the spacecraft experiences only the acceleration due to Martian gravity. The patched conic method uses only one central force for each of the phases and never considers the transition points where two forces attract the spacecraft significantly.

The reason for using patched-conics is to get an approximate ΔV necessary for interplanetary transfers, departures, and captures. Patched-conics provide an initial guess for boundary value problems as will be shown later.

To illustrate how patched-conics work, suppose we have a satellite in a geosynchronous orbit that we wish to place in orbit about Mars (either near-Mars polar[8] or Mars synchronous orbit). Assume also that both Earth and Mars have zero eccentricity orbits, and the orbit of Mars lies in the same orbital plane as Earth.[9] Additionally we will perform a minimum energy transfer, a Hohmann transfer, between the planets.

First, calculate the perihelion and aphelion velocities ($v_{\odot,p}$ and $v_{\odot,a}$) assuming that Mars and Earth are not present. Use the specific energy relations, Eqs (14) and (15)[10]

$$\begin{aligned}
\mathcal{E}_t &= -\frac{\mu_\odot}{r_{\odot,a} + r_{\odot,p}} && (14)\\
\mathcal{E}_t &= \frac{v_\odot^2}{2} - \frac{\mu_\odot}{r_\odot} && (15)\\
r_{\odot,p} &= r_\oplus && (16)\\
&= 149.5 \times 10^6 \text{ km} \\
r_{\odot,a} &= r_m && (17)\\
&= 227.8 \times 10^6 \text{ km}
\end{aligned}$$

to solve for $v_{\odot,p}$ and $v_{\odot,a}$.

$$\begin{aligned}
v_{\odot,p} &= \sqrt{2\mu_\odot\left(\frac{1}{r_{\odot,p}} - \frac{1}{r_{\odot,a} + r_{\odot,p}}\right)} && (18)\\
v_{\odot,a} &= \sqrt{2\mu_\odot\left(\frac{1}{r_{\odot,a}} - \frac{1}{r_{\odot,a} + r_{\odot,p}}\right)} && (19)
\end{aligned}$$

---

[8] The Mars polar orbit is evaluated for an inclination of 92.9 degrees and altitude of 360 km.

[9] In reality, $e_\oplus = 0.0167, e_m = 0.0934$, and $i_m = 0.03229$ rad.

[10] For the Sun, $\mu_\odot = 1.327 \times 10^{11}$ km^3/sec^2.

$$\begin{aligned} v_{\odot,p} &= 32.73 \text{ km/sec} \\ v_{\odot,a} &= 21.49 \text{ km/sec} \end{aligned}$$

Now find the orbital velocities of the planets with respect to the Sun ($v_{\odot,\oplus}$ and $v_{\odot,m}$).

$$\begin{aligned} v_{\odot,\oplus} &= \sqrt{\frac{\mu_\odot}{r_\oplus}} \qquad (20) \\ &= 29.78 \text{ km/sec} \\ v_{\odot,m} &= \sqrt{\frac{\mu_\odot}{r_m}} \qquad (21) \\ &= 24.14 \text{ km/sec} \end{aligned}$$

Time-of-flight is given by $\pi\sqrt{a^3/\mu_\odot} = 258.6$ days.[11] Since this is a Hohmann transfer, the directions of both $\vec{v}_{\odot,p}$ and $\vec{v}_{\odot,a}$ are parallel with $\vec{v}_{\odot,\oplus}$ and $\vec{v}_{\odot,m}$, respectively. The next step is to move into the Earth frame, ignore the other objects, and *patch the conic* $v_{\odot,p}$ into the hyperbolic Earth departure velocity ($v_{\oplus,\infty}$).

$$\begin{aligned} v_{\oplus,\infty} &= v_{\odot,p} - v_{\odot,\oplus} \qquad (22) \\ &= 2.95 \text{ km/sec} \end{aligned}$$

Calculate the orbital radius of a geosynchronous orbit ($r_{\oplus,sync}$) and the orbital velocity ($v_{\oplus,sync}$).[12]

---

[11] The orbit for a Hohmann transfer sweeps out π radian, exactly 1/2 an ellipse's area.

[12] For the Earth, $\tau_\oplus$ = 23h 56m 04s, and $\mu_\oplus = 3.986 \times 10^5$ km^3/sec^2.

$$
\begin{aligned}
r_{\oplus,sync} &= \sqrt[3]{\mu_\oplus \left(\frac{\tau_\oplus}{2\pi}\right)^2} && (23)\\
&= 42164 \text{ km}\\
v_{\oplus,sync} &= \sqrt{\frac{\mu_\oplus}{r_{\oplus,sync}}} && (24)\\
&= 3.07 \text{ km/sec}
\end{aligned}
$$

Now calculate the hyperbolic orbit's perigee velocity ($v_{\oplus,p}$) for the $v_{\oplus,\infty}$ at $r_{\oplus,sync}$.

$$
\begin{aligned}
v_{\oplus,p} &= \sqrt{2\left(\frac{v_{\oplus,\infty}^2}{2} + \frac{\mu_\oplus}{r_{\oplus,sync}}\right)} && (25)\\
&= 5.25 \text{ km/sec}
\end{aligned}
$$

If $\vec{v}_{\oplus,p}$ and $\vec{v}_{\oplus,sync}$ are colinear, then $v_{\oplus,sync}$ can be subtracted from $v_{\oplus,p}$ to find the $\Delta V_\oplus$ necessary. This is the optimum ΔV, since subtracting $\vec{v}_{\oplus,sync}$ from $\vec{v}_{\oplus,p}$ vectorially could result in a larger ΔV. For a particular transfer start date we can calculate the inclination range of the orbit to accomplish the optimum ΔV, however this is not necessarily geostationary.[13] Performing this calculation, the optimum ΔV to leave Earth orbit is:

$$
\begin{aligned}
\Delta V_\oplus &= v_{\oplus,p} - v_{\oplus,sync} && (26)\\
&= 2.18 \text{ km/sec}
\end{aligned}
$$

---

[13] A geostationary orbit has a zero inclination relative to the Earth's equator, not the plane of the ecliptic.

The next step is to move into the Mars frame and find the hyperbolic approach velocity ($v_{m,\infty}$).

$$\begin{aligned} v_{m,\infty} &= v_{\odot,m} - v_{\odot,a} \qquad (27) \\ &= 2.65 \text{ km/sec} \end{aligned}$$

Similar to Eqs (23) and (24), calculate the orbital radius at Mars synchronous orbit ($r_{m,sync}$), and the orbital velocity for both the synchronous orbit ($v_{m,sync}$) and a polar orbit ($v_{m,polar}$) at an altitude of 360 km.[14]

$$\begin{aligned} r_{m,sync} &= \sqrt[3]{\mu_m \left(\frac{\tau_m}{2\pi}\right)^2} \qquad (28) \\ &= 20428 \text{ km} \\ r_{m,polar} &= 360 \text{ km} + R_m \qquad (29) \\ &= 3740 \text{ km} \\ v_{m,sync} &= \sqrt{\frac{\mu_m}{r_{m,sync}}} \qquad (30) \\ &= 1.45 \text{ km/sec} \\ v_{m,polar} &= \sqrt{\frac{\mu_m}{r_{m,polar}}} \qquad (31) \\ &= 3.39 \text{ km/sec} \end{aligned}$$

Now calculate the hyperbolic Mars periapsis velocity ($v_{m,p}$) for the $v_{m,\infty}$ at both $r_{m,sync}$ and $r_{m,polar}$.

---

[14] For Mars, τ_m = 24h 37m 23s, R_m = 3380 km, and $\mu_m = 4.283 \times 10^4$ km^3/sec^3.

$$
\begin{aligned}
v_{m,p,sync} &= \sqrt{2\left(\frac{v_{m,\infty}^2}{2} + \frac{\mu_m}{r_{m,sync}}\right)} && (32)\\
&= 3.35 \text{ km/sec}\\
v_{m,p,polar} &= \sqrt{2\left(\frac{v_{m,\infty}^2}{2} + \frac{\mu_m}{r_{m,polar}}\right)} && (33)\\
&= 5.48 \text{ km/sec}
\end{aligned}
$$

As for the Earth departure, if $\vec{v}_{m,p}$ and $\vec{v}_{m,sync}$ ($\vec{v}_{m,polar}$) are colinear, then $v_{m,sync}$ ($v_{m,polar}$) can be subtracted from $v_{m,p}$ to find the ΔV_m necessary. This is the optimum ΔV since subtracting the vector quantities $\vec{v}_{m,sync}$ from $\vec{v}_{m,p}$ ($\vec{v}_{m,polar}$) could result in a larger ΔV. For a polar orbit, the satellite does not require a plane change for orbit insertion[15]. For the Mars equatorial orbit, however, a plane change may be necessary. Since Mars is inclined at 0.4186 radians with respect to the ecliptic, the maximum plane change necessary will be 0.4186 radians.[16] The range of ΔV values depends on when the satellite arrives.[17]

$$
\begin{aligned}
\Delta V_{m,sync} &= v_{m,p,sync} - v_{m,sync} && (34)\\
&= 1.90 \text{ km/sec (minimum)}\\
\Delta V_{m,sync} &= \left(v_{m,p,sync}^2 + v_{m,sync}^2 - 2v_{m,p,sync}v_{m,sync}\cos(.4186)\right)^{1/2} && (35)\\
&= 2.11 \text{ km/sec (maximum)}\\
\Delta V_{m,polar} &= v_{m,p,polar} - v_{m,polar} && (36)
\end{aligned}
$$

---

[15] The geometry involved for Mars orbit insertion is similar to the Earth latitude restrictions for minimum and maximum launch inclination.

[16] Maximum occurs at summer and winter solstices, and minimum occurs at spring and fall equinoxes.

[17] Additionally, the ΔVs necessary to go from the synchronous orbit to a polar transfer orbit and then to a polar circular orbit are 1.70 km/sec and 1.02 km/sec respectively, for a total ΔV of 2.72 km/sec.

$$= 2.09 \text{ km/sec}$$

As the ideal case shows, transferring from an Earth synchronous to a Mars synchronous orbit requires a total ΔV ranging from 4.08 kn /sec to 4.31 km/sec. The total ΔV for a transfer to a polar orbit directly is 4.29 km/sec. [18] There are many complex scenarios where the eccentricities and relative tilts of the orbital planes of the planets are taken into account. In these scenarios, Venus flybys and non-Hohmann transfer orbits are used. The previous calculations where included to provide approximate numbers, and not to provide the answers to all possible scenarios.

4.2.2.2 Boundary Value Problems (93:117- 120). Patched conic methods have discontinuities that should be eliminated. Using a boundary value problem formulation can eliminate such discontinuities. The 'shooting' boundary value problem requires using an initial state vector at t_1 and propagating forward in time to find the state vector at t_2. One problem with this method is that it requires a decent first guess. The patched conic method provides the decent first guess.

Once an orbit is propagated to t_2, we can test the state vector for how closely it corresponds to a set of desired conditions. If the vector is close, the error will be small; if the vector is not so close, the error will be larger. Suppose the state vector $\vec{X}$ expresses the position and velocity in a rectangular coordinate system centered at the center of mass of our solar system,

$$\vec{X}^T = \begin{bmatrix} x & y & z & V_x & V_y & V_z \end{bmatrix} \tag{37}$$

or alternatively,

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[18] By going to synchronous first and then polar orbit requires a ΔV of 6.80 - 7.03 km/sec.

$$\vec{X} = \begin{Bmatrix} \vec{r} \\ \vec{v} \end{Bmatrix} \tag{38}$$

where $\vec{r}$ is the position vector, and $\vec{v}$ is the velocity vector.

Then a relationship that provides how $\vec{X}(t_2)$ changes with a change in $\vec{X}(t_1)$ can be written as in Eq (39).

$$\delta\vec{X}(t_2) = \Phi(t_2, t_1)\delta\vec{X}(t_1) \tag{39}$$

The matrix $\Phi(t_2, t_1)$ is referred to as the state transition matrix and is usually computed numerically. For Eq (39) to be true, $\Phi(t_2, t_1)$ must be:

$$\Phi(t_2, t_1) \equiv \frac{\partial\vec{X}(t_2)}{\partial\vec{X}(t_1)} \tag{40}$$

$$= \begin{bmatrix} \frac{\partial\vec{r}(t_2)}{\partial\vec{r}(t_1)} & \frac{\partial\vec{r}(t_2)}{\partial\vec{v}(t_1)} \\ \frac{\partial\vec{v}(t_2)}{\partial\vec{r}(t_1)} & \frac{\partial\vec{v}(t_2)}{\partial\vec{v}(t_1)} \end{bmatrix} \tag{41}$$

Substituting Eq (40) into Eq (39) yields:

$$\delta\vec{X}(t_2) = \begin{bmatrix} \frac{\partial\vec{r}(t_2)}{\partial\vec{r}(t_1)} & \frac{\partial\vec{r}(t_2)}{\partial\vec{v}(t_1)} \\ \frac{\partial\vec{v}(t_2)}{\partial\vec{r}(t_1)} & \frac{\partial\vec{v}(t_2)}{\partial\vec{v}(t_1)} \end{bmatrix} \delta\vec{X}(t_1) \tag{42}$$

To calculate the amount that the numerically integrated result differs from the desired conditions, define a vector $\vec{G}$ for the conditions. As an example of a shooting boundary value problem, consider a satellite transfer from Earth to Mars. It is important to point out, you can specify as many end conditions as options available at the departure time. For our example: 1) the desired distance from Mars is the height h, 2) the satellite and Mars lie in the same ecliptic plane, and 3) h is measured at periapsis. This is only one of many possible examples of the end conditions. The conditions at Mars arrival are then as follows (notice that $\vec{G} = \vec{0}$ when the conditions are met):

$$\vec{G} = \left\{ \begin{array}{c} ((x - x_m)^2 + (y - y_m)^2 + (z - z_m)^2)^{1/2} - R_m - h \\ z - z_m \\ (x - x_m)(\dot{x} - \dot{x}_m) + (y - y_m)(\dot{y} - \dot{y}_m) + (z - z_m)(\dot{z} - \dot{z}_m) \end{array} \right\} \tag{43}$$

Defining a matrix B,

$$B \equiv \frac{\partial \vec{G}}{\partial \vec{X}(t_2)} \tag{44}$$

$$B = \left[\begin{array}{cccccc} \frac{x-x_m}{|\vec{r}-\vec{r}_m|} & \frac{y-y_m}{|\vec{r}-\vec{r}_m|} & \frac{z-z_m}{|\vec{r}-\vec{r}_m|} & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 \\ \dot{x} - \dot{x}_m & \dot{y} - \dot{y}_m & \dot{z} - \dot{z}_m & x - x_m & y - y_m & z - z_m \end{array} \right]_{t_2} \tag{45}$$

where,

$$|\vec{r} - \vec{r}_m| = \left((x - x_m)^2 + (y - y_m)^2 + (z - z_m)^2\right)^{1/2}$$

Taking the equations for $\Phi(t_2, t_1)$ and B (Eqs (40) and (44)) and multiplying;

$$B\Phi(t_2,t_1) = \frac{\partial\vec{G}}{\partial\vec{X}(t_1)} \tag{46}$$

Eq (46) specifies how $\vec{G}$ changes with a change in the initial state vector $\vec{X}$, which is exactly what is desired. Allowing only a change in the state vector's initial velocity ($\delta\vec{v}(t_1)$) and fixing the position vector at t_1, results in Eq (47).

$$\delta\vec{G} = \left[\partial\vec{G}/\partial\vec{r}(t_1) \quad \partial\vec{G}/\partial\vec{v}(t_1) \right] \left\{ \begin{array}{c} \delta\vec{r}(t_1) \\ \delta\vec{v}(t_1) \end{array} \right\} \tag{47}$$

Eq (47), which is the product $B\Phi\,\delta\vec{X}(t_1)$ from Eq (46) and the 'δ' form of Eq (38), reduces to Eq (48) since the allowable change in positions is now zero.

$$\delta\vec{G} = \left[\frac{\partial\vec{G}}{\partial\vec{v}(t_1)}\right] \delta\vec{v}(t_1) \tag{48}$$

Solve Eq (48) for $\delta\vec{v}(t_1)$.

$$\delta\vec{v}(t_1) = \left[\frac{\partial\vec{G}}{\partial\vec{v}(t_1)}\right]^{-1} \delta\vec{G} \tag{49}$$

Eq (49) tells you how to change the initial velocity vector such that the condition $\vec{G}$ is reduced to $\vec{0}$.[19] By repeating this process we eventually converge on the actual velocity needed at t_1 to meet the conditions at t_2.

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[19] The $\delta\vec{G}$ in Eq (49) is the change necessary to change the current propagated condition to the desired condition ($\vec{G} = \vec{0}$).

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A complete mission timetable is computed by repeating the shooting boundary value problem at each phase of the mission. For instance, a shooting boundary value problem is performed for each of the following: the launch to LEO transfer, the LEO transfer to a circularized low Earth orbit, the circularized LEO to the Mars transfer orbit, the transfer orbit into a Mars captured orbit, the captured orbit into the operational orbits, and so forth.[20] Each phase would use the final conditions of the previous phase as the initial conditions for the current phase.

The shooting boundary value problem works well for impulsive type ΔVs. However, for longer duration burns, the force has to be included in the computation of $\Phi(t_2, t_1)$, where t_1 would be the start of the burn and t_2 would be the end of the burn. The decision of the mission planner for a low thrust burn is how to change the force such that $\Phi(t_2, t_1)$ produces the desired end conditions.

4.2.2.3 State Transition Matrix Computation(93:106–117). The state transition matrix provides the behavior of nearby orbital trajectories which numerical integration of the state vector alone does not accomplish without having to repeat the entire process.

Numerical integration requires first order differential equations. To integrate the equations of motion numerically, express $\dot{\vec{X}}$ as a function of $\vec{X}$ and t. For example, the ideal two-body problem equations of motion would be as in Eq (51).

$$\dot{\vec{X}} = f(\vec{X}, t) \tag{50}$$

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[20] There are many other phases since the periodic orbits would require propagation from the arrival time until the departure time.

$$\left\{\begin{matrix} \dot{x} \\ \dot{y} \\ \dot{z} \\ \dot{V}_x \\ \dot{V}_y \\ \dot{V}_z) \end{matrix}\right\} = \left\{\begin{matrix} V_x \\ V_y \\ V_z \\ \frac{-\mu x}{(x^2+y^2+z^2)^{3/2}} \\ \frac{-\mu y}{(x^2+y^2+z^2)^{3/2}} \\ \frac{-\mu z}{(x^2+y^2+z^2)^{3/2}} \end{matrix}\right\} \tag{51}$$

Defining the matrix $A(t)$:

$$A(t) \equiv \frac{\partial \dot{\vec{X}}}{\partial \vec{X}} \tag{52}$$

then for the simple two-body problem,

$$A(t) = \begin{bmatrix} 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 \\ \frac{-\mu}{r^3} + \frac{3\mu x^2}{r^5} & \frac{3\mu xy}{r^5} & \frac{3\mu xz}{r^5} & 0 & 0 & 0 \\ \frac{3\mu yx}{r^5} & \frac{-\mu}{r^3} + \frac{3\mu y^2}{r^5} & \frac{3\mu yz}{r^5} & 0 & 0 & 0 \\ \frac{3\mu zx}{r^5} & \frac{3\mu zy}{r^5} & \frac{-\mu}{r^3} + \frac{3\mu z^2}{r^5} & 0 & 0 & 0 \end{bmatrix} \tag{53}$$

where,

$$r = \sqrt{x^2 + y^2 + z^2}$$

The matrix $A(t)$ now tells you how a small change in $\vec{X}$ changes $\dot{\vec{X}}$. Since the state vector $\vec{X}$ can undergo six independent changes $(\delta x, \delta y, \delta z, \delta V_x, \delta V_y, \delta V_z)$, if you simultaneously evaluate the six independent small changes, you will have six different solutions for $\vec{X}$. Define $\Phi(t, t_0)$ as the result of the unit changes of the six independent variables,

$$\dot{\Phi}(t, t_0) = A(t)\Phi(t, t_0) \tag{54}$$

and the initial conditions:

$$\Phi(t_0, t_0) = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 \end{bmatrix} \tag{55}$$

Eq (55), an identity matrix, forces the changes at t_0 to be the changes at t_0. And finally, since each time t can be a new t_0, it is sometimes useful to have the identity:

$$\Phi(t_2, t_0) = \Phi(t_2, t_1)\Phi(t_1, t_0) \tag{56}$$

For the restricted N-body problem where N−1 position vectors are known as a function of time,[21] the equations of motion and $A(t)$ matrix can be computed relatively easily for the remaining object (usually a satellite). The level of detail depends on resources available and accuracy desired.

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[21] Which is common for the motions of heavenly bodies.

To review, with first order equations of motion ($\vec{X}$), find the expression for $A(t)$. Armed with $A(t)$ and $\vec{X}$, numerically integrate both $\Phi(t, t_0)$ in Eq (54) and $\vec{X}(t)$ in Eq (51). The numerically integrated results reveal both the particular solution, $\vec{X}(t_2)$, and how nearby initial conditions affect the particular solution, $\Phi(t_2, t_1)$.

4.2.3 Mission Durations. The data in this section are provided to illustrate typical mission durations and ΔV magnitudes for each of the three trajectories described in Section 4.2.1. These numbers are examples only and are not intended to represent final calculated values.

The values shown in Table 13 are representative of the calculated ΔVs and mission durations associated with the conjunction class missions.[22] These class of scenarios represent Hohmann transfer trajectories and therefore have the smallest overall ΔV magnitudes. However, due to the lower ΔV totals, the mission durations are longer than any other scenario. This aspect is due to waiting for the correct planetary alignment, as discussed in Section 4.2.1. Note the long stays in the Martian system.

Table 13. Earth-Mars Conjunction Class.

Launch Date	ΔV TMI	TOF	ΔV MOI	Mars Stay	ΔV TEI	TOF	ΔV EOI	Mission Length	ΔV Total
mm/yr	km/sec	days	km/sec	days	km/sec	days	km/sec	days	km/sec
04/01	3.6	200	2.5	550	2.1	200	3.4	950	11.6
06/03	3.6	200	2.1	550	2.6	190	3.8	940	12.1
08/05	4.0	220	2.0	490	2.7	210	4.3	920	13.0
10/07	4.2	250	2.0	440	2.3	260	3.9	950	12.4
11/09	4.0	280	2.0	370	2.1	270	4.0	920	12.1
11/11	3.7	250	2.5	420	2.0	260	3.5	930	11.7
01/14	3.8	220	2.8	460	1.9	240	3.5	920	12.0

Table 14 gives the representative values for the opposition class missions on a direct/indirect type trajectory. The outbound leg is very similar to the conjunction class data — a Hohmann transfer. The primary difference in this scenario is the short stay time in the Martian system. Planetary alignment is not a consideration on the return voyage. As described earlier, the indirect portion has an angular change of greater than 180 degrees, and thus results in higher ΔVs for the inbound leg (depending on the particular alignment of Earth and Mars when the return leg begins). Characteristics of opposition class missions are shorter durations but larger ΔV totals.[23]

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[22] The Earth Orbit Insertion (EOI) ΔVs are estimated values and were not included in cited source.

Table 14. Earth-Mars Opposition Class.

Launch Date	ΔV TMI	TOF	ΔV MOI	Mars Stay	ΔV TEI	TOF	ΔV EOI	Mission Length	ΔV Total
mm/yr	km/sec	days	km/sec	days	km/sec	days	km/sec	days	km/sec
03/01	3.6	170	3.6	40	4.8	240	1.4	450	13.4
04/03	4.4	170	4.0	40	4.6	250	1.0	460	14.0
05/06	5.8	180	4.1	40	4.7	260	2.6	480	17.2
08/07	4.9	180	4.8	40	5.3	240	7.9	460	22.9
10/09	4.0	210	4.0	40	5.7	220	9.3	470	23.0
11/11	3.6	250	2.9	40	6.1	210	7.7	500	20.4
01/14	3.6	210	3.4	40	5.4	220	4.5	470	16.9

Table 15. Earth-Mars Opposition Class (with Venus Swingby).

Launch Date	ΔV TMI	Venus Swingby	TOF	ΔV MOI	ΔV TEI	TOF	ΔV EOI	Mission Length	ΔV Total
mm/yr	km/sec		days	km/sec	km/sec	days	km/sec	days	km/sec
04/01	3.6	Inbound	200	2.5	4.2	350	3.6	590	13.9
08/02	3.8	Outbound	300	4.7	3.1	260	3.8	600	15.4
06/04	4.1	Outbound	340	4.4	2.6	270	4.3	650	15.4
08/07	4.6	Inbound	190	4.3	4.0	340	4.3	570	17.2
01/09	4.2	In & Out	330	3.3	3.4	370	4.2	740	15.1
11/10	4.4	Outbound	330	3.5	2.5	300	4.0	670	14.4
11/13	3.7	Inbound	280	2.5	4.4	310	3.8	630	14.4

Use of a Venus swingby improves the ΔV total for the opposition class missions (reference Table 15).[24] The swingby reduces the maneuver requirement at the high-velocity end of the inbound path by removing velocity in the atmosphere of Venus. This makes the short-duration opposition class scenario more competitive with the lower velocity conjunction class trajectories. Note that the mission durations are a compromise between the two methods previously described (57:7–9).

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[23] The indirect/direct opposition class mission is similar to the larger ΔVs occurring on the outbound leg. Mission durations and ΔV totals remain nearly the same.

[24] All mission duration calculations for the Venus swingby have a stay in the Martian system of 40 days.

The ΔVs and mission durations shown here represent the three most likely classes of maneuvers for an Earth-Mars mission. There are many other proposals available, but they are basically derivatives of these three options.

4.2.4 Mission Analysis Summary. The specified mission constraints of one-way trip times less than one year and overall mission durations of under three years are not binding on any of the scenarios presented. The conjunction class mission is the preferred trajectory for an unmanned mission to Mars, because it has the lowest total ΔV required. This reduces the overall mission mass and, thus, the cost. Opposition class missions are more desirable when a manned element is introduced. The faster trip times with these missions reduce the cosmic effects as well as long duration in a microgravity environment. In all cases with opposition class missions, it is desirable to use a Venus swingby on one of the legs.

4.3 Mars Transfer Vehicle

The MTV is the largest factor to consider in the selection of a Mars Transportation System option. The vehicle consists of many subsystems that must be successfully integrated to achieve the desired performance and meet mission requirements. The MTV design is primarily driven by the mission objectives and requirements; however, it cannot operate or be developed in a *vacuum*. It must be designed for integration within the support infrastructure and operate within the constraints of orbital mechanics.

Just as the vehicle is the most critical factor to consider in the selection of the transportation system, the propulsion subsystem is the driving factor in the selection of the MTV. In this section we will analyze various propulsion subsystem options available. The Power, Navigation, Guidance, & Control (NGC), Telemetry, Tracking, Command, & Communications (TTC&C), Structure, Payload Interface, and Thermal Control subsystems will be examined at a requirements level only.

4.3.1 Propulsion. Two factors useful in an analysis of any propulsion system are thrust and specific impulse (Isp). The basic equations used for determining thrust, total impulse, and specific impulse are used extensively in evaluating the propulsion technologies and determining their usefulness based on specific mission scenarios. The specific mission scenarios may be defined in many ways, but the most common are TOF, thrust-to-weight ratio (T/W), and total change in velocity for orbit transfer.

Unfortunately, the optimization of one parameter will sometimes adversely impact another. For example, the reduction of propulsive energy and the reduction of TOF are mutually exclusive goals at this time. The relative importance of these two options will directly impact the type of propulsion to used (77:5).

The T/W required for a particular mission depends on the time allocated for the mission. A high-thrust mission will create a faster acceleration. The penalty for a high-thrust mission is in the increased mass from added fuel. On the other hand, a low-thrust mission will use less fuel, but take longer to accelerate (27:24). Both systems will accelerate a craft; however, the difference in acceleration will change the time required to develop the change in velocity required to leave Earth orbit.

For analysis of specific parameters, several equations are of interest. The basic rocket equation gives the thrust of a rocket:

$$T = m\dot{v} = u_e\dot{m} \tag{57}$$

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where $m\dot{v}$ is force of thrust, $\dot{m}$ is the change of the rocket mass due to propellant loss and u_e is the velocity of the exhaust gas (27:8–15) (47:2).

Specific Impulse is often used as a measure of engine efficiency. In general, Isp is a property of the working fluid used to provide acceleration and may be determined from the enthalpy[25] of the fluid in the engine chamber (H_c) and the exit nozzle (H_e). This relationship may be expressed as:

$$Isp = \sqrt{\frac{2J}{g}(H_c - H_e)} \tag{58}$$

where J is defined as the mechanical equivalent of heat, and g is the acceleration due to gravity (17:5). As a calculated quantity, Isp is defined by:

$$Isp = \frac{T}{\dot{m}g} \tag{59}$$

$$Isp = \frac{\dot{m}u_e}{\dot{m}g} \tag{60}$$

$$Isp = \frac{u_e}{g} \tag{61}$$

where g is defined as the acceleration due to gravity. From these equations, it is evident that Isp and u_e differ by a factor of 10 since $g \approx 10$ m/sec^2 (27:4).

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[25] Enthalpy: Sum of the internal energy of a body and the product of its volume multiplied by the pressure.

The equation for Isp provides useful insights into mass requirements for the vehicle. For a given thrust, thrust time, and engine Isp, the mass of the expelled propellant may be found from:

$$m_p = \frac{Tt}{gIsp} \tag{62}$$

where m_p is the mass of expelled propellant and t is the time of the engine burn (27:19) (47:4).

Also of concern is the change in velocity required to leave the Earth's orbit. The maximum change in velocity may be estimated with the following equation:

$$max\Delta V = u_e ln(\frac{m_o}{m_f}) \tag{63}$$

where m_o is the initial mass of the vehicle and m_f is the mass of the vehicle after the propellant is burned.

By recognizing m_f is the mass of the rocket after the propellant burn, it is possible to relate the max ΔV to the amount of propellant used (m_p). Since $m_f = m_o - m_p$,

$$\Delta V = u_e \ln \frac{m_o}{(m_o - m_p)} \tag{64}$$

solving for m_p gives,

$$m_p = m_o(1 - \exp(-\frac{\Delta V}{u_e})) \tag{65}$$

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Since u_e is related to Isp, it is also possible to solve for m_p in terms of Isp (69:207).

$$m_p = m_o(1 - \exp(-\frac{\Delta V}{gIsp})) \tag{66}$$

For the purpose of our mission, m_o may be defined as:

$$m_o = m_{p/l} + m_{pws} + m_p + m_t \tag{67}$$

where $m_{p/l}$ is the mass of the payload and cradle assembly, m_{pws} is the mass of the power system and subsystems, and m_t is the mass of the propellant tanks. For mass calculation purposes, propellant tanks are assumed to be 14 percent of m_p (31:4).

Equation (66) indicates the utility of a large Isp. For a given requirement in ΔV, the propellant mass fraction m_p/m_o decreases as Isp increases. However, research has shown that the sacrifice for high Isp is increased mass on the propulsion system. For a Nuclear Thermal Propulsion (NTP) system, this mass is the reactor and engine assembly. However, for a Nuclear Electric Propulsion (NEP) system, the propulsion mass includes the reactor, the radiation shielding, the excess heat rejection system, and the thrusters. The need or desire for a high Isp and, therefore, a high u_e and low m_p must be tempered with considerations of the power plant weight needed to drive the engine (47:7).

4.3.1.1 Chemical Propulsion. The use of chemical[26] propulsion has been the traditional mainstay of the space program. As interplanetary missions become more frequent and larger in mass, the limitations of this propulsion system become more evident. The relatively low Isp of the current state of the art engines requires massive amounts of propellant to achieve the required change in velocities. As a result, the Initial Mass in Low Earth Orbit (IMLEO) of chemical propulsion systems are driven by propellant requirements (77:2).

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[26] Sometimes referred to as cryogenic because of the super cold nature of the fuel and oxidizer.

The chemical rocket uses the energy stored in the chemical propellant to heat a working fluid and expand it through a nozzle (77:2). Based on its formulation, Isp depends on the exit velocity of the propellant (u_e). It can be shown that it is advantageous to use a propellant with a low molecular weight in order to get the largest possible u_e (and therefore Isp) (42:353–355). This is the reason why hydrogen (H_2) is so attractive as a propellant. However, the choice of H_2 creates some problems in long term storage. For long term missions, a percentage of the propellant must be assumed lost to boil off — two percent is often used (31:4).

Unfortunately, u_e is limited by the heat generated in the chemical oxidation process. At the current state-of-the-art, Isp for a chemical rocket is limited to the range of 300-500 seconds (17:8) (42:371). This inefficient use of propellant requires more of the total vehicle mass be allocated to propellant to achieve the same results as some newer technologies.

When compared to electric propulsion, chemical systems have much greater thrust and shorter burn times. However, the system also has a much lower Isp when compared to NEP due to the differences in u_e. Additionally, a chemical propulsion system poses technical problems for future aerobrake missions at Mars because of possible excess speed on arrival (77:12). For a pure aerobrake descent, it is possible a Venus swing by would be required to lower the arrival velocity (77:15).

In general, fewer options are available with the use of chemical propulsion. Chemical systems have varying energy requirements and it is hard to design for vehicle commonality. The heavier the payload, the less attractive this option becomes (77:19). As a result, chemical propulsion was not competitive as a possibility for this mission because of the lack of growth potential.

4.3.1.2 Nuclear Propulsion. The use of a nuclear power source provides advancements in both thermal and electric propulsion. As a heat source, it can heat propellant to much greater chamber temperatures than a chemical reaction. As a result, Isp improves by a factor of at least two. As a Multi-Megawatt (MMW) power source, the reactor provides the necessary power to run the electric thrusters, through power conversion techniques. While advantages are evident, these advantages come at a price in terms of propulsion system total mass.

Nuclear Reactor Designs. The past power requirements for spacecraft engines were on the order of 1 to 10 kW; however, the current requirements for space exploration places the requirement into the hundreds of kW to MMW range. To solve this problem, nuclear reactor design research started in 1956 with the Aircraft Nuclear Propulsion Program. This program was later extended to nuclear rocket engines[27] and Space Nuclear Auxiliary Power (SNAP). As a result of these and other research efforts, there are several reactor designs considered viable for space propulsion systems (65:406).

The liquid-metal-cooled, solid-core design is an outgrowth of SNAP research. This reactor underwent continuous development from 1956 to 1972. The reactor operated at an efficiency of six percent and produced 35 kW. This particular reactor has growth potential into MMW ranges and could be used to provide power to electric propulsion systems as well as other onboard subsystems (65:407).

The gas-cooled, solid-core reactor began as part of the Rover research and culminated in the Nuclear Engine for Rocket Vehicle Application (NERVA) designs of 1972. The full scale development reactor was approaching flight-rated status when the program was cancelled. This reactor was designed to provide 890 kN of thrust as the propellant was passed through the reactor to both cool the reactor and provide heat to the propellant. The Isp of this design was 925 seconds (31:3). While the

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[27] Project Rover - NASA program that investigated nuclear propulsion technology from the late 1950s to the early 1970s.

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initial designs experienced vibration problems, these problems were solved prior to program cancellation (65:408–411).

Fixed-particle-bed and rotating-particle-bed reactors both make use of pelletized fuel elements. The small, individual elements increase surface area for heat transfer and as a result raise the Isp of the propulsion system. The recently announced Timberwind Program makes use of this technology with a reported Isp of 1,000 seconds (5:18–20). Other sources report an Isp of 1,050 seconds with this design (31:4). This reactor also has the possibility of increasing the power to that of electric propulsion systems by incorporating liquid metal cooling and a power conversion system.

The gas-core nuclear reactor has the greatest potential for future development since it is capable of operation at temperatures above the melting point of all known metals. The high heat of the reactor provides the maximum possible heat transfer to the propellant passing through the reactor. However, a major problem with the design is the loss of gaseous fission products through the nozzle. Current research is directed toward finding a way to contain the fission products in a mineral cavity capable of withstanding the heat. In this way, the propellant could pass by without removing the reactor fuel (65:418). The possible Isp for this design could reach 2080 seconds(36:6).

Nuclear Thermal Propulsion. Nuclear Thermal Propulsion uses a nuclear reactor to heat the propellant. As a result, the heat transfer is much greater than could be achieved through chemical oxidation of the propellant. As a general rule, the Isp of a NTP will be on the order of twice the Isp of a chemical rocket; therefore, for a given payload, the IMLEO may be reduced significantly through savings in propellant mass (73:1).

The renewed interest of the government in NTP was demonstrated when ***Aviation Week & Space Technology*** revealed the existence of the Timberwind Project. This project is a follow on to the NERVA studies initiated in 1956 and cancelled in

1972 when Mars exploration missions were put on hold. Although the study used a particle bed reactor, NASA reportedly did not want to commit to a specific reactor design (5:18–20). However, since Timberwind reportedly tested a reactor which could provide up to 75,000 lb thrust with an Isp of 1,000 seconds, the Synthesis Group[28] strongly recommended NTP be considered in the baseline system for Mars exploration (6:38) (5:18–20).

Nuclear Electric Propulsion. Early in the history of space exploration, scientists recognized the utility of electric propulsion systems where the trip would extend over months or years (84:vii). NEP uses a MMW reactor to create a voltage potential used to ionize a propellant. The ions are accelerated and expelled to generate the needed thrust. The MMW power level is a requirement to accelerate these particles. The high-power level requires extensive shielding and heat rejection systems (77:2–4). Additionally, there is a requirement for a power conversion system. The IMLEO of NEP is driven by the mass of these systems (77:5).

Nuclear Electric Propulsion is characterized by a long spiral time to escape a gravity well, high Isp, low thrust, and high payload mass fraction. As a rough order approximation, acceleration levels of NEP are on the order of .001 m/sec^2. As a result, thrusting is done as a long continuous burn to accelerate and to decelerate the spacecraft (77:2).

Thrusters. There are several possibilities for thruster configurations in the NEP system. These thrusters may be grouped into three general areas differentiated by their method of propulsion: electrostatic, electromagnetic, and inducted pulsed plasma. Each of these areas uses the MMW power of the reactor, but applies that power in different ways.

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[28] The "Synthesis Group" refers to commission chaired by Astronaut Thomas P. Stafford, USAF Lieutenant General (Ret), which authored the report *America at the Threshola* and is synonymous with "Stafford Commission."

Electrostatic thrusters use a gaseous propellant. An element such as mercury, argon or xenon is ionized by electron impact to form a neutral plasma. A high voltage potential is then applied to the ionized gas to extract the ions from the plasma. Using electrostatic forces, the ions are then accelerated to extremely high velocities into an ion beam. For electrostatic thrusters, Isp is an independent design variable which may be specified by the accelerating voltage. Ranges of Isp are from 1,500 to 10,000 seconds. Additionally, thruster efficiency is directly proportional to Isp. Based on a demonstrated thruster efficiency of 50–80 percent, the maximum thrust per module ranges from one μN to one Newton. The greatest advantage of this technology is that it has been flight tested (69:214–215).

Electromagnetic thrusters use magnetic forces to accelerate the propellant from the engine. The current research status of these thrusters is varied. Two of the current designs are teflon pulsed plasma and magnetoplasmdynamic (MPD). The teflon plasma thruster operates by passing a large voltage through a gas to create a magnetic field. This field then ablates and ionizes a portion of the teflon propellant block. The ionized gas is then accelerated by gasdynamic forces and ohmic heating[29]. This system has several advantages: no propellant storage requirements, no pumping problems, and only two voltage sources are required (69:216–218).

Researchers have noted MPD test data is hard to get because of the requirement for large pumping capacity. Current research is focusing on pulsed research which will give the best chance for continuous operations. If this research is successful, Isp will range between 1,000 and 4,000 seconds (69:218- 220).

Inducted Pulsed Plasma uses an electric pulse to create a magnetic field from a plasma. The magnetic field then accelerates a piece of propellant to high velocity. The advantage of this system is that any solid object may be used. The Isp range of this system is 1,000 to 2,000 seconds but with an efficiency of less than 50 percent.

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[29] Ohmic heating - Heating due to an electrical current through a resistor.

Additionally, to be competitive as a propulsion system, this system must be capable of greater than 10^7 cycles (69:220- 222).

Heat Rejection. Heat rejection is also an issue for NEP since the reactor is not cooled by the propellant gas as in nuclear thermal propulsion. This portion of the system with the reactor and shielding provide the largest contribution to mass in the NEP option. There are several radiation designs being considered in current research. All of these designs operate by rejecting excess heat to the space environment; however, there are several design factors which contribute to the overall size and mass of the radiator. These factors include rejection temperature, deployable or fixed configuration, and meteor protection (3:101–108). Current predictions estimate that for a 10 MW power source the radiator mass would be 1,580 kg (20:457).

Power Conversion. For the NEP system there must be a way to convert the thermal energy of the reactor into usable electric energy. This process is accomplished through use of a power conversion system. These systems operate on direct transfer of heat to electric power or on the principles of thermodynamics. The direct conversion technique is known as thermionics.

Thermionics uses the high temperature of the reactor to boil off electrons from an emitter across a gap to a cooler interelectrode.[30] The first generation of in core thermionics was tested in the early 1970's and efforts were directed toward producing 120 kW. Current research is directed toward out of core thermionics which allow greater efficiency in both the reactor and the conversion system (3:93–96).

Currently, there are three thermodynamic cycles applicable to space power system. These cycles are Brayton, Rankine, and Stirling. All of the cycles operate on the principle of compression and expansion of a working fluid. Unlike thermionics, extensive machinery is required to accomplish the power conversion (3:75-88), and thus are potentially more massive than thermionic systems. A representative mass breakout by reactor electric power is given in Table 16 (20:457).

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[30] An anode to cathode eletron emission.

Table 16. Nuclear Reactor Masses.

Mass	1 MW	5 MW	10 MW
	kg	kg	kg
Reactor	860	2310	3620
Primary Loop	290	330	390
Shielding	880	1300	1800
Power Converter	875	1400	2200
Radiator	621	935	1580

4.3.1.3 Solar Electric Propulsion. SEP propulsion works just like NEP in that electric forces are used to generate propulsion. The same thruster technology used for NEP would be applicable to SEP, except that the method used to generate power is different. For SEP, a massive solar array collects energy from the Sun and transfers it directly into electricity.

Current studies show that 10 MW of power may be produced; however, this power level requires two massive solar arrays measuring 140m x 145m each. The solar array with its associated support structure could have a mass as great as 129.8 mt. To gain optimum performance, studies have shown that thrusters should be designed for an Isp of 5,000 seconds (43:5).

The size and complexity of the solar arrays creates severe construction challenges and complexities for on-orbit assembly. Additionally, the vehicle could experience up to a 30 percent decrease in array performance when the craft passes through the Van Allen radiation belts (43:5). These complexities combined with the lower Isp indicate SEP has only limited usefulness in the accomplishment of our mission objectives.

4.3.1.4 Summary. Analysis of the four propulsion systems reviewed indicates two systems, NTP and NEP, stand out as options for Project Ares. NTP provides greater thrust, greater acceleration, and reduced TOF; however, the lower Isp of NTP requires more total vehicle mass be reserved for propellant. Both systems have advantages and disadvantages which must be weighed in relation to Mission Analysis and the Support Infrastructure.

4.3.2 Power. The MTV will require electrical power during all phases of the mission and must be capable of providing the required electrical power to mission payloads during the Earth-to-Mars transfer. Power requirements for the MTV are estimated to be less than 500 W. This level should be sufficient to run on-board subsystems, including NGC, TTC&C, and Thermal Control. Maximum power requirements of Phase II mission payloads while in transit are estimated at 1,500 W. (reference Section 3.8). This power level should be sufficient to keep the payloads powered up, but not functioning, to keep temperature sensitive components from freezing.

Since future payload requirements are unknown, a 50 percent power system margin is added to give a maximum total electrical power requirement of approximately 3 kW. This power must be provided between the Trans-Martian Injection (TMI) burn and satellite separation near Mars. After satellite separation,the power requirement would drop back down to the 500 W required by the MTV.

Independent of the type of power system used, the MTV must carry rechargeable batteries for use during emergency conditions. The mass of these batteries are extremely sensitive to the exact required power levels and operating periods. As this report looks only at the development of the mission and MTV subsystems requirements, and not at a complete design of the MTV and its subsystems, the required battery mass cannot be calculated. For a real design, the exact load profiles during emergency conditions would have to be determined. Generally, for nuclear propulsion systems requiring batteries for emergencies, silver-zinc are the best choice due to their low mass. Chemical or solar propulsion systems using solar power systems would also need batteries for solar eclipse periods. Generally, nickel-hydrogen batteries would be the best choice due to their greater cycle life (23:124).

The electrical power system is not a driving parameter for the MTV design. The power system design is driven almost entirely on the type of propulsion system the MTV uses. If the MTV uses nuclear electric propulsion, the MMW reactor driving the propulsion unit will easily meet any additional parasitic loads. If the MTV uses solar electric propulsion, the MMW solar arrays will easily cover any parasitic loads.

If the MTV uses nuclear thermal propulsion, small sets of thermoelectric or thermionic converters can be placed near the reactor to generate the required electrical energy. For thermoelectric devices, a high temperature heat source is required to make a solid-state semiconductor device produce electricity. For themionic devices, a high heat temperature heat source is required to make electrons boil off a tungsten cathode and flow to an anode. Thermoelectric efficiencies typically run from four to five percent, while thermionic devices operate between 10 and 20 percent. Compared to the mass of the reactor core, both conversion devices would have negligible mass. Thermoelectric devices have been used on all of the deep space probes, and they have a very good reliability and lifetime records. For that reason, thermoelectric devices are the better choice for a NTP system.

If the MTV uses chemical propulsion, a band of solar cells mounted on a cylindrical surface (such as the fuel tanks) will generate the required electrical power. At Mars aphelion, the solar constant is 488 W/m^2 (reference Section 2.2 and 3.8). For a gallium arsenide solar panel operating at 28°C and pointing directly at the Sun, a cross sectional area of about 45 m^2 will be sufficient (23:107). For a cylindrical shape five meters in diameter with the broadside 30 degrees off normal from the Sun, this cross sectional area can be achieved using a band of solar cells according to the following equation (1:347).

$$2\,r\,w = \text{cross section/off pointing loss} \qquad (68)$$

where:

r = rocket body radius = 2.5 m
w = width of solar cell band around rocket body
cross section = 45 m^2
off pointing loss = cos 30°

This results in a width of about 10.4 m. These solar cells must be mounted on a rigid surface. If a cylindrical surface is available on the MTV, then the additional mass required by the solar cells is negligible. If an appropriate surface is not available, rigid solar array panels mounted in the required cylindrical configuration will cover 163 m^2 and mass approximately 500 kg (1:345). This mass is virtually negligible compared to the total MTV mass.

4.3.3 Navigation, Guidance, and Attitude Control. A basic requirement for effective operation of the MTV is the ability to follow a precise trajectory in a prescribed vehicle orientation with respect to Earth, Mars, and inertial space at all times during a mission. The system which accomplishes this is the Navigation, Guidance, and Control (NGC) Subsystem. Of particular importance in NGC subsystem's component selection is the requirement for autonomy of hardware and software operations due to the nature of interplanetary operations.

4.3.3.1 Navigation. Navigation is the determination of the current state of motion of the MTV. The state of motion consists of position, velocity, and attitude. This state is determined with reference to some mission-dependent coordinates

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suitable for defining the motion of the MTV (25:5–1). The challenge of navigating in space results from the lack of gravity which on Earth allows for the establishment of a local vertical.

4.3.3.2 Guidance. Guidance is the process of 1) comparing this measured navigation state with a required MTV state and 2) computing commands to correct differences between the two (25:5-4). The required MTV state depends on the particular mission requirements. Attitude measuring sensors for guidance include gyroscopes, accelerometers, Earth/Mars/Sun sensors, and star trackers.

The primary guidance instrument is the gyroscope. A gyroscope is an instrument that employs a rapidly spinning mass to sense the inertial orientation of its spin axis, i.e. angular position. Rate gyroscopes and rate integrating gyroscopes are attitude sensors used to measure changes in the MTV's orientation (23:199).

One of the most recent developments in the area of gyroscopic instruments is the laser gyroscope. In this device two beams of light follow a triangular path in opposite senses using reflecting mirrors. If the device is rotating about an axis perpendicular to the light path, the inertial path followed by the light beam moving against the rotation will be slightly shorter than the path length followed by the light beam moving in the same sense as the rotation (92:165).

The path lengths each way will differ from each other by approximately one part in one billion—a minute quantity to measure. As a result, monochromatic laser light is employed. Using light of only one wavelength, the two beams combine with each other to produce interference patterns on the surface of each mirror. These interference patterns are then measurable.

A ring laser gyroscope is currently under development for NASA's next generation of planetary spacecraft. In this technology, the light path is over four km of optical quartz fiber wound on a spool. The large path length provides greater sensitivity, without the saturation problems at high angular rates common in the mechanical rate gyroscope. Another advantage over the mechanical rate gyroscope is its lack of moving parts (92:166). We have selected the ring laser type of gyroscope for the MTV.

During ΔVs, the MTV is subjected to forces caused by the thrust of the propulsion system. These forces are measured by accelerometers which provide guidance information and assist in keeping maneuvers within specified limits. The accelerometer is a device which measures applied forces. The MTV includes a force-measuring component composed of three accelerometers mounted along orthogonal axes in order to measure forces in all three directions (71:138).

To determine velocity and distance traveled, the NGC subsystem includes integrating accelerometers. The integrating accelerometer provides an output proportional to the velocity gained owing to a force acting on the MTV.

Earth and Mars sensors on the MTV are used to scan across the respective planet, measuring rotation angles to define the spacecraft's attitude relative to the planet (23:179). In planetary orbit, the respective planet presents an extended target to a sensor when compared to the Sun and stars due to their relative distances from the MTV. Planets have a relatively constant blackbody exitance in the infrared wavelength region. This radiation is detected and measured using a sensor consisting of a scanning mechanism, an optical system, a detector, and signal control electronics.

The MTV has Sun sensors to detect the Sun as a point source. The Sun can be used as a reference when the MTV is in planetary orbit, since the orbital altitudes of the MTV are much smaller than the distance from the vehicle to the Sun (23:188).

Star trackers are the final attitude measuring equipment required by the MTV. Star tracking sensors allow the MTV to obtain relatively fixed references during the interplanetary flight phase of a mission. These sensors use an optical telescope to focus the star image on a photodetector. The location of this image is then compared to the known location of the star and a correction factor is computed.

4.3.3.3 Attitude Control. Attitude control is the application of corrective maneuvers by active and passive means to obtain changes commanded by guidance (25:5-1). There are forces existing in the space environment which attempt to upset the required attitude of the MTV. As a result, it is necessary to provide attitude sensing devices and control systems to maintain the desired attitude. The following are factors in the space environment which cause disturbing forces to act on the MTV:

- Atmospheric drag.
- Gas molecules and micrometeoroids.
- Gravitational gradients.
- Magnetic and electrostatic fields.
- Solar wind and radiation pressure.
- Uncompensated motion of internal moving parts.

The factors listed above vary in degree according to the MTV's current state: either planetary orbit (Earth or Mars) or interplanetary travel. Vehicle attitude stabilization requires a combination of passive and active measures. During the journey to Mars, the MTV will be spin-stabilized, but once in planetary orbit, the vehicle will be fixed in three axes with respect to the planet.

Spin stabilization is a passive technique in which the MTV acts as a gyro wheel with a high angular momentum. Radiation-pressure stabilization is a second passive technique that is practical for interplanetary flight where the gravitational fields of the planets may be neglected. The radiation pressure exceeds the effects of the solar gravity gradient at the distance of the Earth's orbit (71:150).

Active attitude stabilization techniques must also be employed due to the size of the MTV with payload. One active technique used is gas-jet/thruster control. Gas-jet control involves an expulsion of mass in one direction with gas jets to overcome an upsetting force in the opposite direction. The system consists of a tank containing gas under high pressure, a regulator, control valves, and gas jets for each axis: yaw, pitch, and roll. The thrust of the gas produces a torque about the center of mass which overcomes the disturbing torque (71:151).

The specific impulse of a cold gas system is rather low, on the order of 50 seconds. This would not be useful due to the MTV's size. Higher impulses are obtained by using hot gas monopropellants such as hydrogen peroxide or bipropellants. The system is similar to the Reaction Control System of the STS in which 44 small engines are used for minor attitude adjustments on orbit. This system has the capability to be refueled for multiple missions.

The mass of an attitude control subsystem is a function of 1) the type of attitude stabilization, 2) required attitude control accuracy, 3) redundancies in actuators and sensors, and 4) size and mass of the spacecraft. The complexity and mass of an attitude control subsystem increases with the increase in attitude control requirements and size of the spacecraft (1:49). Based upon data from Earth-orbiting satellites, the mass estimate for the NGC subsystem would be on the order of 1,000 kg, composed primarily of propellant for the thrusters. Power requirements are relatively small on the order of 50 W.

4.3.4 Telemetry, Tracking, Command, and Communications. During mission operations, the status of the MTV's subsystems and payload require monitoring by the Mars Mission Control Center (MMCC). Any deviations from the nominal status must be corrected, either by autonomous error-correcting components or by command from mission controllers. This requirement is satisfied by the TTC&C subsystem which provides telemetry downlink and commanding services of the MTV in order to conduct ground monitoring and control of the spacecraft and its payload.

4.3.4.1 Telemetry. Telemetry for the MTV consists of measurements taken by vehicle and payload sensors (transducers) which are transmitted to an Earth ground station. Vehicle sensors monitor the state of health of the transportation system. Payload sensors monitor the status of mission systems aboard the MTV.

A sensor is a device that measures the physical parameter of position, pressure, temperature, or radiation and converts them into an electrical signal (23:236). The measurement of the vehicle and payload local environment aids in determining the state of individual subsystems and can be used for failure diagnosis. Thus, the telemetry system collects the necessary data and transmits it in a particular format.

In the case of a large number of sensors on the MTV, it is necessary to multiplex, or combine, their outputs for transmission by one or more transmitters. The outputs of the sensors are then properly amplified, multiplexed, and broadcast over the communication system.

The telemetry system employs various modulation schemes. One of these is Pulse Code Modulation/Frequency Modulation/Phase Modulation (PCM/FM/PM). In this scheme, the PCM Encoder Unit encodes analog and digital information from various subsystems by means of time division multiplexing. It performs analog-to-digital conversion and organizes the information into a serial bit stream whose frequency modulates a voltage-controlled oscillator and finally phase modulates a transmitter (23:239).

4.3.4.2 Tracking. Tracking involves the location of the MTV in time and space and the MTV's motion as a function of time (25:5–21). Purposes of tracking include 1) allowing for the transmission of commands to the MTV, 2) acquiring telemetry, and 3) providing data for interplanetary flight path determination.

Due to the nature of interplanetary flight, optical tracking is not applicable. As a result, the MTV is tracked using radar tracking and ranging. The tracking system on the vehicle consists of a transponder that is triggered when the Earth-based tracking radar transmits. This system has extensive range and accurate range measurements (23:251).

4.3.4.3 Command and Communications. Commanding is the method of controlling the MTV from Earth while the vehicle remains in the line of sight of a ground station. Communications with the MTV are accomplished using NASA's DSN[31]. Commands are transmitted to the MTV upon direction from the MMCC. Two types of commands may be sent to the vehicle: real-time and stored program. The MTV receives and reacts to real-time commands immediately. Stored program commands activate MTV systems and sensors while the vehicle is out of the LOS for DSN communication (25:5–21).

The MTV communications system consists of receivers, transmitters, and corresponding antennas. Communication antennas on the MTV are steerable feedhorns which can be continuously controlled to ensure that they are always directed toward the Earth. The usual operations frequency for uplink transmission is the 6 GHz; for downlink transmission, 4 GHz. Since wider bandwidths are possible at higher frequencies, the 16 GHz and 36 GHz bands will be considered with the added advantage of avoiding ground network interference (23:255).

The TTC&C subsystem is composed of transmitters, receivers, antennas, transponders, modulation equipment, and associated electronics. Based upon data from Earth-orbiting satellites, the mass estimate for the TTC&C subsystem is 200 kg (23:261–266). Power requirements are primarily those of the transmitters. The power estimate for the TTC&C subsystem is 350 W (71:215).

4.3.5 Structure. The structure is the backbone of any space vehicle which ties together all other subsystems of a spacecraft. Basic structural design requirements for the MTV include (23:73-79):

- *Support and Structural Integrity.* The spacecraft structure is the primary load-bearing component designed to withstand the most severe combination of loads and vibrations possible in any phase of the mission. Some of these loads include: steady state and dynamic acceleration launch loads; shocks due to payload, fairing, and stage separations; vibrational loads due to noise (engine, buffeting, and boundary layer); and pressure loads.

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[31] The DSN was covered in Section 2.1.

- *Structural Interface.* The structure will provide primary structural interface for both the initial launch vehicle and mission payloads.
- *MTV Support.* The structure will be designed to facilitate on-orbit support of assembly, payload mating, and refurbishment activities.
- *Safety Factors.* Safety factors will be incorporated into structural design to cover uncertainties in design, load limits, and operational spectrum. Safety factors must be designed, keeping in mind that increased safety factors result in corresponding increases in vehicle mass.

The structure of the MTV will depend heavily on the physical configuration and placement of all other subsystems and payloads. Our analysis addresses only the development of the mission and MTV subsystems requirements, not a complete design of a MTV. Additionally, launch vehicle (assumed to be the NLS) quasi-static loads and vibrations levels are unknown; therefore, a detailed structural discussion on load and vibration limits, primary and secondary stiffness requirements, and design safety factors, is not possible at this level. The mass of a spacecraft's structure accounts for five to twenty percent of the spacecrafts launch weight (23:88).

4.3.6 Payload Interface The MTV is an all-purpose vehicle that will carry many different types of payloads throughout this and the following phases of Project Ares. The MTV is not designed to carry a specific payload; rather, the payloads will be designed for integration with the MTV. The payloads that the MTV will carry, by virtue of their different missions, are vastly different in design and construction. It is not possible to use the same structural design for an orbiting communications satellite as for a mobile surface laboratory or surface habitat module. These differently designed and constructed spacecraft will experience vastly different forces in the various flight regimes of the mission and will require different types of structural interfaces between the MTV and the primary structure of the payload.

One solution to suppporting a highly flexible structural interface is to incorporate a payload structure adaptor or cradle assembly that is custom-built to carry the particular payloads of a mission. The cradle assembly is constructed to provide stable support and maximum structural integrity to the payload through the payload's primary structure attach points, while having a generic interface (attach points) with the MTV. This approach is not new. The STS, for example, utilizes this method to carry a variety of payloads. NASA's Mission Particular Equipment Support Structure and various cradle assemblies are mounted in the Shuttle's cargo bay using standard keel, trunnion, and longeron fittings (23:84-85).

An electrical interface is also required. This interface provides required electrical power from the MTV's power system to payloads along with an electrical ground connection. The interface also provides telemetry and command links to the payloads through the MTV's TTC&C subsystem as well as command links to the payload cradle assembly release mechanism. This electrical interface utilizes a double umbilical connection between the payloads and the MTV. The first umbilical connects the payload to the cradle assembly—a standard umbilical connection used today. It would disconnect upon payload separation from the cradle assembly. The second umbilical connects the payload cradle assembly to the MTV. This is a more robust connection, since it has to support a variety of payload power, telemetry, and command requirements. It requires an in-flight disconnect capability to allow the separation of the payload cradle assembly as well as simple connection operation to aid in the mating of the payload cradle assembly to the MTV.

The mission payloads are mated to the payload cradle assembly prior to launch. The generic structural and electrical interface would also aid in the integration of payload cradle assembly to launch booster. A shroud surrounds the payload cradle assembly to protect the payload during launch. After reaching orbit, the MTV, its payload cradle assembly, and the shroud are separated from the booster. The payload shroud is later released at the time dictated by mission requirements.

As stated in the previous section, structure mass typically accounts for five to twenty percent of a spacecraft's total initial mass. For the purpose of MTV mass calculation, a payload cradle assembly mass of 15 percent of the payload mass will be used. Given a maximum payload mass of 6.5 mt, this brings the total payload and cradle mass to 7.5 mt.

4.3.7 Thermal Control. The operation of any mechanical or electrical piece of hardware will be effected by temperature, resulting in an anomalous operation or complete failure if the temperature is outside a particular range. For spacecraft, these ranges vary depending on the spacecraft design as well as the particular subsystem. The following are representative subsystem *allowable operational temperature* ranges (23:300):

- TT&C system: – 5 to 50 degrees centigrade.
- Antenna: – 5 to 70 degrees centigrade.
- Power system: 0 to 60 degrees centigrade.
- Batteries: 0 to 20 degrees centigrade.
- Attitude control system: 0 to 60 degrees centigrade.

Heat can be generated internally by the electronic equipment and externally by absorption of solar and planetary albedo energy. Areas of the spacecraft generating or absorbing too much thermal energy can force the temperature to exceed upper operating limits of particular subsystems in that area. On the other side of the spectrum, areas of the spacecraft not generating or absorbing enough thermal energy can cause the temperature to drop below the lower operating limit. The spacecraft's thermal control system maintains the temperature of all spacecraft subsystem equipment within their designed operational ranges.

There are two basic types of thermal control: active and passive. Passive thermal control methods control temperature by using materials with various conductive and radiative properties to direct and insulate the flow of thermal energy into and out of specific areas of the spacecraft. Some common types of passive thermal control methods are: thermal coating materials, insulation, and heat sinks. System designers will consider the following factors in the design and use of passive thermal control systems:

- *Thermal Properties.* The emmissivity and absorptivity of a materials used in passive thermal control will determine how efficient the material is as a radiator, absorber, or conductor.
- *Material Degredation.* Most materials used for thermal coating degrade with time, exposure to solar radiation, and other man-made contaminants (23:305).
- *Limitations.* Passive methods are limited in that their effectiveness decreases as the range of temperature variation increases (23:305).
- *MTV Integration.* Use of passive thermal control methods must be examined with respect to their impact on MTV mass and other MTV subsystems.

Active thermal control methods pick up where passive systems leave off. Active methods involve the monitoring of spacecraft subsystem equipment temperatures. Upon reaching temperature limits, active thermal control hardware, such as heaters, thermal louvers, heat pipes, or active cooling systems are turned on or off to adjust the temperature. System designers will consider the following factors in the design and use of active thermal control systems:

- *System Failure.* Active thermal control systems are mechanically and/or electronically operated and, therefore, subject to failure. Redundancy and positive control of these systems are essential in any design.
- *Autonomous Operations.* Due to the maximum communication time delay of approximately 40 minutes, in addition to the various other periods of MTV communication inaccessibility (solar conjunctions, planetary eclipses, et cetera), continuous realtime monitoring of subsystem equipment temperatures will be impossible. Systems will be designed for autonomous and/or preprogrammed operation.
- *MTV Integration.* Use of active thermal control methods must be examined with respect to their impact on MTV mass and other MTV subsystems, including the power they will require for operation.

The specific design of the thermal control system, as with the structure of the MTV, will depend on the physical configuration and placement of all other subsystems and payloads. We evaluate only the developmental aspects of the mission and MTV subsystems requirements, not the complete design of a MTV. A detailed thermal analysis of the MTV to examine specific types and placement of thermal control devices is not possible. This research will be conducted as part of an actual MTV design process. The mass of the thermal control subsystem has already been included in the mass of the MTV structure.

4.3.8 Vehicle Requirements Summary. There are basically four MTV options available—each differing by the type of propulsion system employed. The propulsion system forces the type of power subsystem used for each vehicle type. The remaining subsystem requirements remain as they were previously described.

In an effort to examine the effect of Isp and propulsion technology on the different MTV options, equation (66) and (67) were solved to determine first order estimates of the variables $m_{p/l}$, m_t, m_{pws}, and m_p. To enter the equation, at least

Table 17. MTV Mass: Conjunction Mission – Current Technology

Mass Required	Chemical Isp =500	NTP–NERVA Isp = 925	NEP Isp = 5000	SEP Isp = 5000
	mt	mt	mt	mt
m_{pws}	59.8	40.6	196.1	67.0
$m_{p/l}$	7.5	7.5	7.5	7.5
m_t	96.3	24.4	9.5	3.5
m_p	642.2	162.6	63.3	23.2
IMLEO	805.8	235.1	276.4	101.1

one of the variables had to be known. Since data was available on the general mass of the power system and subsystems, m_{pws}, this number was used (79:6).

Using the m_{pws} data and the Isp values referenced in the propulsion section, masses for each configuration under study were calculated and a first order estimate of the IMLEO was determined. Data in Tables 17 through 21 show that Isp and power system mass significantly impact the final IMLEO. IMLEO is also significantly effected by the type of orbital transfer used. In order to get an estimate on the upper bound of IMLEO, masses were calculated for the ΔV requirements for both conjunction and opposition class missions. For the conjunction mission (reference Table 13), the August 2005 mission was used with a total $\Delta V = 13$ km/sec. For the opposition class mission (reference Table 15), the August 2007 mission was used with a total $\Delta V = 17.2$ km/sec.

Mass calculations in Tables 17 through 20 assume a reusable vehicle returning to Earth. The chemical system listed in these tables is based on the maximum limit of Isp possible as referenced in the propulsion section. Table 21 gives mass requirements for an expendable chemical MTV. This IMLEO is based on the assumption that all components of the MTV will be discarded after a one-way trip to Mars. A first order analysis indicates that there are tremendous savings in IMLEO by discarding the MTV after payload deployment; however, the costs of vehicle development, construction, and deployment currently offset the benefit of using one-way vehicles.

Table 18. MTV Mass: Opposition Mission – Current Technology

Mass Required	Chemical Isp =500	NTP-NERVA Isp = 925	NEP Isp = 5000	SEP Isp = 5000
	mt	mt	mt	mt
m_{pws}	59.8	40.6	196.1	67.0
$m_{p/l}$	7.5	7.5	7.5	7.5
m_t	286.0	46.5	13.4	4.9
m_p	1906.7	309.7	89.1	32.6
IMLEO	2260.1	404.3	306.0	111.9

Table 19. MTV Mass: Conjunction Mission – Improved Technology

Mass Required	Chemical Isp =500	NTP-PBR Isp = 1000	NEP Isp = 10000	SEP Isp = 5000
	mt	mt	mt	mt
m_{pws}	59.8	33.6	196.1	67.0
$m_{p/l}$	7.5	7.5	7.5	7.5
m_t	96.3	17.3	4.3	3.5
m_p	642.2	115.3	28.9	23.2
IMLEO	805.8	173.7	236.8	101.1

Table 20. MTV Mass: Opposition Mission – Improved Technology

Mass Required	Chemical Isp =500	NTP-PBR Isp = 1000	NEP Isp = 10000	SEP Isp = 5000
	mt	mt	mt	mt
m_{pws}	59.8	40.6	196.1	67.0
$m_{p/l}$	7.5	7.5	7.5	7.5
m_t	286.0	31.7	5.9	4.9
m_p	1906.7	211.1	39.3	32.6
IMLEO	2260.1	283.9	248.8	111.9

Table 21. MTV Mass: One Way Trip – Chemical Propulsion

Mass in mt	Conjunction Mission Isp =500	Opposition Mission Isp = 500
	mt	mt
m_{pws}	59.8	59.8
$m_{p/l}$	7.5	7.5
m_t	42.5	64.2
m_p	283.4	427.9
IMLEO	393.3	559.5

From strictly a total mass perspective, the SEP option would be the best. It has a IMLEO of at least 70 percent less than the next closest option; however, its size and complexity, degradation of solar arrays, and Isp limit all serve to decrease its low mass benefit. The chemical MTV, including the one way trip option, is limited in its maximum Isp and has a significantly greater mass than other options.

Of the remaining NTP and NEP options, the NEP option has much less thrust, less acceleration, and greater TOF. The thrusters of a NEP system, however, make efficient use of the propellant. This efficiency translates into increased payload mass. NEP will take considerable more time to leave and enter a planet's gravity well than the NTP, essentially having to spiral in and out at low thrust until escape velocity is reached. TOF differences between NTP and NEP is on the order of two to three months.

Both the NEP and NTP will require on-orbit construction. From a size point of view, it appears the NTP will be less challenging to construct. The NTP designs currently under study show a NTP attached to a central support assembly. Propellant tanks are attached to and within the support structure. NEP designs require the additional construction of a radiation shield and thermal heat radiators. Assuming the support and construction infrastructure is already in orbit, the NTP would be easier to assemble.

The biggest challenge facing a NTP design is the long-term storage of H_2 propellant. Current models of Mars missions require the assumption of propellant bleed off during the trip (31:4). This problem only occurs if one desires to recover and reuse the NTP transfer vehicle or use propulsive capture into Mars orbit as our systems have been modeled. As of April 1991, NASA officials were complaining that no funding had been allocated to perform research on long-term cryogenic fuel storage (5:74–75). If this storage problem is solved, it would make the NTP a more competitive option. Unlike NTP, NEP uses a heavy substance like mercury for propellant. Our research revealed no information concerning storage problems in any of the reviewed simulations.

Both the NEP and NTP MTV options are competitive options that meet mission requirements and objectives. The selection of a propulsion system will be made with respect to its integration with the Mission Analysis and Support infrastructure.

4.4 Support.

The logistics support required for the MTV is the final factor to consider prior to making a recommendation for a transportation system. In examining the support required by the various options of MTVs, consideration must be given to the probable growth of the selected MTV design to support the increased payload requirements of future phases of Project Ares. The two main areas of support that need to be considered are 1) support necessary to get the required hardware into LEO and 2) support required while in LEO.

4.4.1 Earth-to-Orbit. The ETO mass requirements for Phase II of Project Ares alone, are greater than any other single project ever undertaken. The availability of heavy lift launch vehicle to support the program is required before the program can even begin; therefore, we assume that the NLS is operational and can support this program.

The size, and in particular, the mass of the MTV is a very important factor in the selection of the overall transportation system from a support perspective. The cost of initially placing MTVs and required refurbishment mass into LEO could account for up to 30 to 50 percent of the total life cycle cost of the entire Mars program (43:7). To examine the ETO requirements for each MTV option, the total ETO mass requirements for each option is examined.

For comparison purposes, we analyzed MTV masses (excluding payload mass) for the conjunction class transfer trajectory using current technology engines. ETO mass requirements from Tables 19 and 21 are summarized in Table 22. Initial assembly mass consists of total IMLEO minus payload mass. Refurbishment mass includes replacement propellant and tank masses.

Table 22. MTV ETO Mass Requirements

	Chemical Direct	Chemical Reusable	NTP	NEP	SEP
	mt	mt	mt	mt	mt
Initial Assembly	385.8	798.3	227.6	268.9	93.6
Refurbishment	325.9	738.5	187.0	72.8	26.7

The selected MTV option will be used for the two payload delivery flights in Phase II. Assuming its use in two more payload delivery flights in Phase III, a four flight program can be used as a baseline for life cycle MTV ETO mass requirements. These requirements are summarized in Table 23.

Table 23. Life Cycle MTV ETO Mass Requirements

	Chemical Direct	Chemical Reusable	NTP	NEP	SEP
	mt	mt	mt	mt	mt
Initial Assembly	385.8	798.3	227.6	268.9	93.6
Refurbishment	977.7	2215.5	561.0	218.4	80.1
Total	1362.9	3013.8	788.6	487.3	173.7

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Examining the ETO mass requirements for the various MTV options, the following conclusions can be noted:

- SEP MTV has the lowest initial and lifecycle IMLEO.
- Chemical MTV has the highest initial and lifecycle IMLEO.
- NTP MTV has a lower inital IMLEO than the NEP option; however, due to the higher Isp of the NEP, refurbishment masses are much lower. Consequently, lifecycle IMLEO is lower for the NEP as compared to NTP.
- These IMLEO mass differences will be increased as Isp increases and lifecycle of the MTV increases.

Another consideration is that the selected MTV option will be used not only for Phase II payload delivery, but also to validate the concept needed for the development of the larger scale MTVs necessary to support future phases. A review of Equations (66) and (67) shows the exponential relationship between payload mass, total vehicle mass, and the mass of required propellant. From these equations it can be seen that as payload requirements increase, total vehicle mass increases linearly, and propellant mass increases exponentially. This increase is greater for the Chemical and NTP MTVs with large propellant requirements, than the NEP and SEP MTVs. Simply put, the IMLEO advantages of NEP or SEP will increase as vehicle mass increases.

4.4.2 On-Orbit. As with ETO support, on-orbit support required by the MTVs will also be greater than any previous operation, and will be as great a concern in the selection of a transportation system as ETO support. It is assumed that a support infrastructure is currently in place on SSF that can support the assembly, testing, payload mating, launch, maintenance, refurbishment, and refueling of a MTV.

There are other on-orbit support considerations besides the availability of SSF that may be important in the selection of a MTV. They include radiation effects, assembly time and complexity, and refurbishment time and complexity.

4.4.2.1 Radiation Effects. A prime concern of using a nuclear propulsion system is the effect of radiation. If reactors of any type are used on the MTV, they must be launched *clean* and *cold*. This means no fission products are present. The reactor will not start up until just prior to beginning of the TMI burn. After the reactor is brought up to full power, fission products will build up inside the fuel rods. The total amount of fission products generated will exponentially increase as the reactor continues to run, and will saturate at an equilibrium level after several months of operation. While the reactor is operating, the radiation from fission products is negligible compared to the neutron and gamma flux created by the fission process. After the reactor shuts down, the neutron flux vanishes, but the fission products (and other radiation activation processes) will make the reactor permanently radioactive. This level of radioactivity depends on the amount of fission products present, which depends on the operating period length.

NEP systems require an almost continuous burn from Earth to Mars with the reactor operating for many months. Due to the low thrust of the NEP, it will need to spiral out of the Earth's gravity well. This time is on the order of three months (77:2). During this period, the reactor will be operating at full power. The high level neutron and gamma flux due to the fission process will be hazardous to manned systems within a certain range. The high radiation flux from the MTV will also damage unprotected satellite systems within a lesser range. This radiation level decreases by the square of the distance away from the reactor. Payloads carried by the MTV will be adequately protected behind the radiation shield which protects a volume along the MTV's long axis.

Due to the radiation flux, operations near SSF will be impossible without additional massive shielding. The geometric problems involved in maneuvering the MTV down to LEOs while avoiding all other satellites will be extremely difficult. All maintenance and refurbishment activities must be performed with specialized robotic equipment. With these factors in mind, it would be best to remotely service the NEP MTV in an isolated orbit, thus, requiring the development of a completely separate telerobotic-supported infrastructure.

For the NTP concept, the reactor would only operate for several hours during each interplanetary trip. The NTP's thrust is high enough that it only remains in LEO for a short time; therefore, the radiation dose to manned systems and nearby satellites is much less than that from the NEP system. This concept makes it easier to bring the NTP MTV into LEO without unintentionally damaging other satellites and allows the performance of maintenance and refurbishment activities within a serviceable range from the on-orbit support infrastructure already established at SSF.

4.4.2.2 Assembly and Refurbishment Considerations. All of the MTV options under consideration have a mass less than 200 mt, excluding propellant. The chemical and NTP systems are more streamlined, and could be placed into orbit on a single launch vehicle (assumed to be the NLS) with no assembly required. The SEP and NEP, while massing less than 200 mt, will require on-orbit assembly due to the large solar array and thermal radiator. When considering the growth of the vehicle to support future phases of Project Ares, however, the MTV masses much greater than 250 mt are possible. For this reason, consideration must be given to the assembly time and complexity involved with the different MTV types.

Assuming that the support infrastructure necessary for each MTV option is in place, NEP and SEP would require approximately 50 percent more time for initial assembly than NTR and Chemical options. This is primarily due to the size and increased complexity of these systems as compared to NTR or Chemical (43:11). Refurbishment turnaround times would be slightly higher for Chemical and NTP due to the greater ETO propellant mass requirements of those vehicles (43:11).

Just as lifecycle ETO mass requirements must be examined, life cycle on-orbit operation times must be reviewed. Actual assembly and refurbishment times would be greatly dependent on the actual design of the MTV. One report assess the higher initial assembly time of the NEP and SEP vehicles as tending to offset the higher refurbishment times of the NTR and Chemical vehicles producing approximately equal life cycle on-orbit operations times (43:11).

4.4.3 Summary. From a support standpoint, the best option appears to be either the Chemical or the NTP MTV. The NEP and SEP systems are much more complex and require greater initial assembly time. Additionally, NEP could not take advantage of the existing support facilities in place on SSF. Due to radiation effects, the NEP would require the establishment of the complete support infrastructure. NTP and Chemical systems have approximately the same initial assembly and refurbishment times and about the same level of complexity. Both systems would take advantage of the existing support infrastructure.

4.5 Transportation System Summary.

Given the scope and objectives of Project Ares, the NTP and NEP MTV options were the most competitive. Given the specific Phase II requirements, the optimal transportation system appears to be a NTP MTV utilizing a low energy conjuction class trajectory and supported in LEO by the existing infrastructure available with SSF.

The choice of transfer trajectory was the easiest to make. A simple comparison of the mass required for the different types of trajectories showed that the low energy (low ΔV) conjuction class trajectories were the most optimal. The TOFs for this class of trajectory are well within the Phase II specific requirements of an Earth-Mars transit time of less than 12 months and a round trip time of less than 40 months. For our unmanned missions, there was no need to waste mass, which translates to money, by going faster.

MTVs used in future phases of Project Ares may have a need to reduce TOFs. One reason might be to take advantage of the 26-month synodic period by launching a mission each period, as opposed to every other period as in Phase II. This could be necessary to support the greater requirements of a permanent manned base on Mars. These possibilities, while considered, had no impact on our selection. Phase II mission requirements and a reusable MTV concept validation can be performed utilizing the lower energy transfers.

The selection of the vehicle type was more difficult. The chemical propulsion MTV was eliminated due to its mass and lack of growth potential. Chemical propulsion systems currently approach the theoretical Isp limit of 500 seconds. This low specific impulse indicates their inefficient use of propellant and thus large propellant mass requirement when compared to the other options. IMLEO estimates show the Chemical MTV to be at least three times as massive as the other options. This large mass was unnecessary to meet Phase II mission requirements. Chemical propulsion systems do, however, provide the high thrust required for high energy, short TOF missions desirable for manned missions. A chemical MTV utilizing aerobraking to reduce propulsive ΔV requirements, and thus mass, may be a competitive option for a manned rated vehicle and should be examined in future phases.

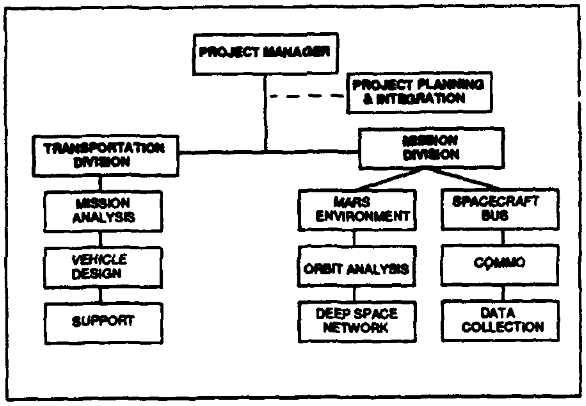
In general, the low thrust NEP and SEP systems would perform the mission the best. Their high Isp, low IMLEO, and low refurbishment masses made them an attractive option. The SEP was selected for two primary reasons: 1) the degradation of the solar array performance would limit lifetime and 2) the lower optimal limit of the system's Isp gave less growth potential. The NEP is a prime candidate for a cargo transportation system. Although it has a high initial assembly mass (due to the massive reactor), the NEP has the most efficient use of propellant mass and, thus, a very low propellant mass requirement and the lowest life cycle mass requirement of any option. The radiation effects expected from the system, however, require the development of a separate support infrastructure which is unwarranted at this point in the project.

The NTP option offers the best combination of all systems. The NTP has a thrust capability approximate to that of chemical systems. By using a nuclear reactor to heat the propellant, the NTP can attain a greater efficiency, and thus reduces propellant requirements. As seen in the NERVA program, NTP systems are a proven technology. Although it uses a nuclear reactor, the NTP can still utilize the current support infrastructure with certain safety precautions. Engine efficiencies are not limited; consequently, the NTP has the potential for growth to support future phases of Project Ares. NTP systems also have the thrust capability for high-energy short TOF missions desirable for manned interplanetary travel.

Although a man-rated MTV was not one of the Phase II requirements, it will be required for Phase IV. Man-rating a current system, or modifying a current system to be man-rated, has obvious advantages over developing a new one. The primary advantage is maximum use of the existing development, test, and support infrastructure. Development of a NTP system for Phase II may have no effect on the future development of the man-rated systems. If the decision is made to use a NTP system for Phase IV manned missions, most of the work will have already been accomplished.

The NTP is the optimal system at this time. All major systems at some point become outdated and are eventually replaced as new systems and technologies mature. Also, project objectives and requirements will continue to evolve. In the future, Project Ares may have expanded to warrant or even require the establishment of a separate "Mars Transportation Space Depot" utilizing a large number of massive MTVs. A NEP-type system may then be the optimal system. By that time, a new propulsion technology may be discovered or significant advancements may be made in current technologies. The bottom line is although systems tend to be designed for maximum use over a given lifetime, requirements and technologies change! As this occurs, the system currently in use may no longer be optimal. There is a continuous need to examine requirements and compare them with available options.

Appendix A. *Project Ares Team Organization*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PROJECT PLANNING & INTEGRATION
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MISSION DIVISION
MISSION ANALYSIS
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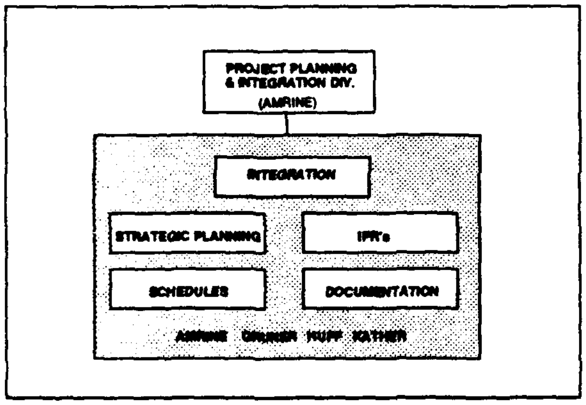
SUPPORT
MARS ENVIRONMENT
ORBIT ANALYSIS
DEEP SPACE NETWORK
SPACECRAFT BUS
COMMO
DATA COLLECTION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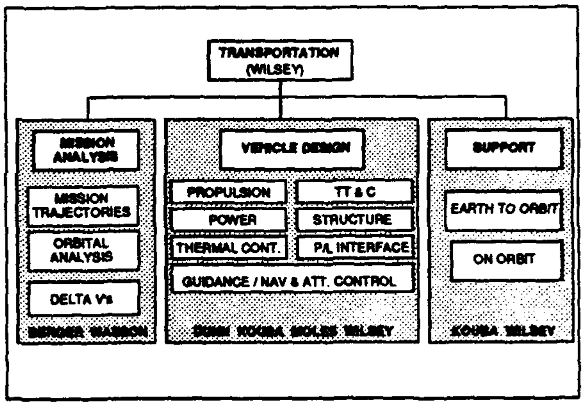
STRATEGIC PLANNING
IPR's
SCHEDULES
DOCUMENTATION
AMRINE GRUNER HUFF KATHER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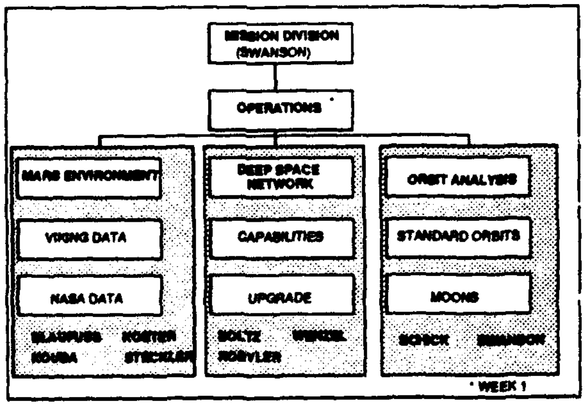
MISSION TRAJECTORIES
ORBITAL ANALYSIS
DELTA V's
BERGER SWANSON
VEHICLE DESIGN
PROPULSION
TT & C
POWER
STRUCTURE
THERMAL CONT.
P/L INTERFACE
GUIDANCE / NAV & ATT. CONTROL
DUNN KOUBA NOLES WILSEY
SUPPORT
EARTH TO ORBIT
ON ORBIT
KOUBA WILSEY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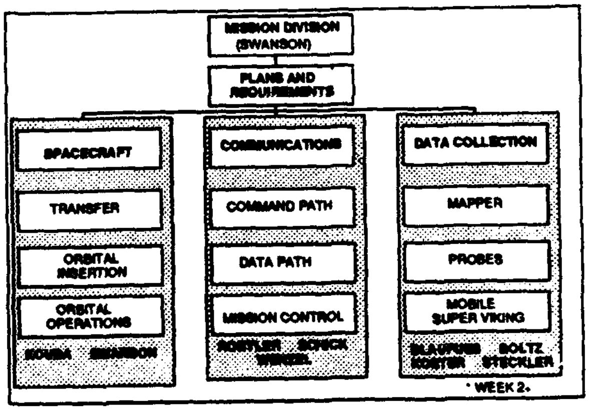
OPERATIONS *
MARS ENVIRONMENT
VIKING DATA
NASA DATA
BLAUFUSS KOSTER KOUBA STECKLER
DEEP SPACE NETWORK
CAPABILITIES
UPGRADE
BOLTZ WENZEL ROEVLER
ORBIT ANALYSIS
STANDARD ORBITS
MOONS
SCHICK SWANSON
* WEEK 1


MISSION DIVISION (SWANSON)
PLANS AND REQUIREMENTS
SPACECRAFT
TRANSFER
ORBITAL INSERTION
ORBITAL OPERATIONS
KOUBA SWANSON
COMMUNICATIONS
COMMAND PATH
DATA PATH
MISSION CONTROL
ROESTLER SCHICK WENZEL
DATA COLLECTION
MAPPER
PROBES
MOBILE SUPER VIKING
BLAUFUSS BOLTZ KOSTER STECKLER
* WEEK 2+

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Appendix B. *Project Ares Phase II Team Members*

Captain John M. Amrine
Captain Jeff M. Berger
Captain David J. Blaufuss
Lieutenant Richard W. Boltz
Captain Michael T. Dunn
Captain Jeffrey S. Gruner
Captain Benjamin C. Huff
Major George R. Kather, USA
Captain David N. Koster
Lieutenant Eric T. Kouba
Captain Joseph B. Moles, USA
Captain Dwight A. Roblyer
Captain William G. Schick, CF
Lieutenant Benjamin T. Steckler
Captain David E. Swanson
Captain Michael S. Wasson
Captain Richard A. Wenzel
Captain David G. Wilsey

Glossary

Ar Argon

AU Astronomical Unit

ACS Attitude Control System

BOL Beginning-of-Life

CCD Charge Coupled Device

C&DH Command and Data Handling

CO Carbon Monoxide

CO$_2$ Carbon Dioxide

CPU Central Processing Unit

DIPS Dynamic Isotope Power System

DSCC Deep Space Communications Complex

DSC-EGA Differential Scanning Calorimeter-Evolved Gas Analyzer

DSCS III Defense Satellite Communications System III

DSN Deep Space Network

DSS Deep Space Station

EOI Earth Orbital Insertion

EOL End-of-Life

EP Experiment Package

ETO Earth-to-Orbit

FM Frequency Modulation

FOV Field-of-View

GCF Ground Communications Facility

GRS Gamma Ray Spectrometer

HIRES High Resolution

IMLEO Initial Mass in Low Earth Orbit

IR Infrared

Isp Specific Impulse

JPL Jet Propulsion Laboratory

LEO Low-Earth-Orbit

LOS Line-of-Sight

LST Local Satellite Time

LTD Lift-to-Drag

MAG/ER Magnetic/Electron Reflectometer

MEDRES Medium Resolution

MEM Microscopic Examination Module

MHA Material Handling Assembly

MMCC Mars Mission Control Center

MMW Multi-MegaWatt

MOI Mars Orbital Insertion

MOLA Mars Observer Laser Altimeter

MPD Magnetoplasmadynamic

MSM Mars Surface Mapper

MTV Mars Transfer Vehicle

NASA National Aeronautics and Space Administration

N_2 Diatomic Nitrogen

NEP Nuclear Electric Propulsion

NERVA Nuclear Engine for Rocket Vehicle Application

NGC Navigation, Guidance, and Control

NO Nitric Oxide

NTP Nuclear Thermal Propulsion

O Atomic Oxygen

PCM Pulse Code Molulation

PM Phase Modulation

PMIRR Pressure Modular Infrared Radiometer

PPM Parts per Million

PROM Programmable Read-Only Memory

RAM Random Access Memory

Rf Radio Frequency

RS Radio Science

RTG Radioisotope Thermal Generator

SDS Sample Distribution System

SEI Space Exploration Initiative

SEP Solar Electric Propulsion

SEU Single Event Upsets

SNR Signal-to-Noise Ratio

SSF Space Station Freedom

STS Space Transportation System

TDRSS Tracking and Data Relay Satellite System

TEI Trans-Earth Injection

TES Thermal Emission Spectrometer

TMI Trans-Martian Injection

TOF Time of Flight

TT&C Telemetry, Tracking, and Commanding

TTC&C Telemetry, Tracking, Command, and Communication

T/W Thrust-to-Weight

TWTA Traveling-Wave-Tube Amplifier

VLBI Very Long Baseline Interferometry

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